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AUTOMATED MARS SURFACE SAMPLE RETURN MISSION CONCEPTS FOR ACHIEVEMENT OF ESSENTIAL SCIENTIFIC OBJECTIVES

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16. Abstract

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PREFACE

The National Aeronautics and Space Administration will continue its exploration of Mars beyond the Viking '75 missions. One of the most exciting, challenging, and potentially worthwhile options being considered is a sample return mission. In 1972 the NASA Administrator directed that a study be performed to determine concepts compatible with a sample return mission which would emphasize essential science only. Emphasis would be on a mission which would not require sample transfer in Mars orbit, and the Titan III E/Centaur would be considered as the baseline launch vehicle.

The study was conducted jointly by personnel from the NASA Langley Research Center (LRC) and the Jet Propulsion Laboratory (JPL), California Institute of Technology from October 1972 to May 1973. Contributions to this report by members of the study team are as follows:

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SUMMARY

Mission concepts were investigated for automated return to Earth of a Mars surface sample adequate for detailed analyses in scientific laboratories. The minimum sample mass sufficient to meet scientific requirements was determined. Types of materials and supporting measurements for essential analyses are reported. A baseline trajectory profile was selected for its low energy requirements and relatively simple implementation, and other profiles were studied for comparison. Trajectory profile design data were developed for 1979 and 1981 launch opportunities. Efficient spacecraft systems were conceived by utilizing existing technology where possible. Systems concepts emphasized the 1979 launch opportunity, and the applicability of results to other opportunities was assessed. It was shown that the baseline missions (return through Mars parking orbit) and some comparison missions (return after sample transfer in Mars orbit) can be accomplished by using a single Titan III E/Centaur as the launch vehicle. All missions investigated can be accomplished by use of Space Shuttle/Centaur vehicles.

INTRODUCTION

Scientific progress and interest in Mars exploration has continued at an accelerated rate since the early years of the National Aeronautics and Space Administration. Already maps of the Martian surface are approaching lunar detail and the Viking mission in 1975 will provide in situ surface sample analyses and preliminary biological information. The USSR's Mars exploration program has been vigorous and persistent; two spacecraft went to Mars in 1971 and four in 1973. Beyond the USSR and Viking '75 missions, the next logical step would be the return to Earth of a sample of the Martian surface.

Return of samples from the planets and satellites is important to the understanding of the solar system and the processes by which it evolved. The question of the existence and nature of life also is difficult to answer conclusively without sample analysis in Earth-based laboratories or in equivalent automated laboratories.

Whether life is found on Mars by Viking or USSR spacecraft, a sample return mission can be expected to answer many questions about Mars which cannot be obtained remotely (particularly with current and near-term technology). For example, accurate age dating will require a returned sample. Experience with lunar samples has demonstrated the necessity for returning samples to Earth-based laboratories – even without the existence of life. If life is found on Mars, returned samples will be required to answer the multitude of questions to be asked by the scientific community. At some point in the exploration of a planet such as Mars, it becomes more cost effective to return a sample to existing Earth laboratories than to continue to build larger and more complex systems for in situ analysis. As an example, the geologist on Earth goes out in the field and does some preliminary analyses, but for detailed analysis he brings the sample back to his laboratory. Preliminary and, in fact, rather complex analyses will be performed in situ by automated systems. However, the types of analyses which provide sufficient details of the current state of Mars and its past evolution (and answers the many questions raised by the in situ analysis) will require the return of a sample to the sophisticated laboratories of Earth. The great merit of working with samples returned to Earth is that the process is parallel and multidisciplinary, and it is adaptive in a way that no laboratory, manned or unmanned, on Mars can be.

In 1970, the Northrop Corporation performed a Mars surface sample return (MSSR) mission study with very ambitious objectives. These objectives included returning large amounts of Martian material, extensive in situ scientific and engineering measurements, and the incorporation of large surface-roving vehicles. To meet these objectives would require systems which would be massive, complex, and costly. As an example, one of the missions which was defined would return a 63-kg sample, but would require a Saturn V with a nuclear stage as the Earth launch vehicle. In 1971, a study by Canady and Darnell at the NASA Langley Research Center developed MSSR missions which could be accomplished by using two Titan III E/Centaurs as Earth launch vehicles. The first vehicle would launch the Mars lander (which would include a Mars ascent vehicle) to a direct entry at Mars, and the second would launch an orbiter (which would include the Earth return vehicle) to be inserted into Mars orbit for later rendezvous and docking with the ascent vehicle. These latter missions, although less ambitious than the former, were developed to return an 11-kg sample and incorporated small rovers and other fairly extensive scientific instrumentation. These studies showed that in planning an MSSR mission, substantial advantage can be gained by employing technology which is already developed.

The study reported herein concentrated exclusively on MSSR missions which acquire and return to Earth Martian samples adequate for scientific investigation. Efforts were concentrated on meeting only the most essential scientific objectives. The sample mass and type and the measurements necessary to support the sample are reported. Acquisition and control of the sample are assumed to be the only science-related functions. Rovers and ancillary scientific instrumentation were not considered. A description of possible options for science enhancement can be found in reference 1.

Trajectory modes for a baseline mission were selected on the basis of minimum energy requirements and simplicity in implementation. Alternate modes which offer flexibility in system design were studied for comparison with the baseline missions. Trajectory design data were developed for 1979 and 1981 launch opportunities.

Systems design and analyses concentrated on the 1979 launch opportunity, and concepts are detailed to a greater depth for the baseline missions than for comparison missions. Systems analyses included techniques employed in reference 2 for minimizing total systems mass. Commonality between systems and current technology were used to the greatest extent possible. Design inheritance from the Pioneer, Mariner, and Viking flight systems, and the development of new designs only where essential improve mission confidence.

Mission concepts investigated included consideration of protecting Mars from contamination by Earth organisms. Most of the systems design requirements for Martian quarantine can be met by using techniques developed for the Viking mission. It should be noted that the problem of protecting Earth from Mars organisms (back contamination) is recognized as a fundamental scientific and engineering issue associated with MSSR missions. The scope of this study, however, allowed only cursory consideration of back contamination. Several mission approaches to the back-contamination problem are suggested in reference 1, and one of those approaches was implemented in the study of this report.

This report treats sequentially the various trajectory phases from Earth departure through Earth arrival. Total launch mass for all missions considered is summarized and Earth launch vehicle requirements are defined. Requirements are assessed for compatibility with the Titan III E/Centaur and Shuttle/Centaur launch vehicles. Study results are discussed in terms of their applicability to other launch opportunities, and overall conclusions of the study are presented.

Some of the study results, with emphasis on systems for the baseline missions, are presented in reference 3, and a companion paper to reference 3 investigated the feasibility of an automated rendezvous and docking and sample transfer in a Mars orbit (ref. 4).

SYMBOLS

A	reference area
a_l	aerodynamic loading during entry

C_3	geocentric injection energy
C_D	drag coefficient
D_o	diameter of parachute canopy
g	planetary gravitational acceleration
h	altitude
h_a	apoapsis altitude
h_d	altitude at parachute deployment
h_p	periapsis altitude
I_{sp}	specific impulse
K	ratio of inert propulsion mass to expended propellant mass
L/D	lift-drag ratio
M_d	Mach number of parachute deployment
M_E	mass at entry
M/C_DA	ballistic coefficient
P_{IQ}	maximum probability of planetary impact in the absence of a final heliocentric trajectory correction maneuver
Q	integrated heating, $\int \dot{q}\, dt$
q	dynamic pressure
$\dot{q}$	heating rate
q_d	dynamic pressure for parachute deployment

T/M	initial thrust-mass ratio of Mars ascent vehicle
t	time
V_d	velocity at parachute deployment
V_E	velocity at entry
$V_{h,a}$	hyperbolic excess velocity at planet arrival
$V_{h,d}$	hyperbolic excess velocity at planet departure
V_1	velocity of first-burn phase of two-burn vehicle
ΔV	velocity increment
α_{trim}	trim angle of attack during entry
γ_E	relative flight-path angle at entry
$\Delta\gamma_E$	uncertainty in flight-path angle at entry
ρ_s	atmospheric density at Mars surface

Subscripts:

C	convective
max	maximum
R	radiative
1,2,. . .	denotes specific version or variant of major spacecraft component

NOMENCLATURE

A/S	Mars entry aeroshell
B/S	Earth bioshield

CMOS	complementary metal oxide semiconductor
COSPAR	Committee on Space Research
DHAC	data handling and command
DBV	dual-burn vehicle
DGB	disk gap band
E - 6	six hours prior to atmospheric entry for direct entry mode
EEC	Earth entry capsule
EOC	Earth orbit capsule
ERV	Earth return vehicle
MAV	Mars ascent vehicle
MCM	Mars cruise module
MEC	Mars entry capsule
MLM	Mars lander module
MOR	Mars orbit rendezvous
MOV	Mars orbit vehicle
MSSR	Mars surface sample return
P/S	Mars parachute system
RTG	radioisotope thermoelectric generator
TDI	terminal descent initiation
TSV	two-stage vehicle

SCIENCE REQUIREMENTS

The scientific value of a sample returned from Mars is unquestioned. A wide variety of measurements that would be extremely difficult or impossible to make remotely on Mars can be made in terrestrial laboratories. Multiple techniques would be available for detailed chemical and biochemical analyses, detection of life, paleontology, isotopic analysis, age dating, mineralogy, petrology, and measurements of physical properties. This section discusses the science requirements for satisfying the objectives of the present MSSR mission (that is, to acquire and return to Earth a sample adequate for essential scientific analyses). Information reported includes (1) quantities and preferred types of materials, (2) techniques and conditions for acquisition of materials, (3) scientific and engineering measurements, and (4) recommendations for sample handling and preservation. These requirements are based on the investigation reported in reference 1. Reference 1 also discusses benefits that could be gained by increasing science capabilities substantially beyond those required to meet the objective of the present missions.

QUANTITIES, TYPES, AND PARTICLE SIZES OF SAMPLE MATERIALS

In recommending sample masses and particle sizes, it was assumed that primary interests would center in the biological, chemical, and geological fields. Of prime importance are detection of life forms (present or past), age dating, chemical compositions, and petrology. The minimum sample masses and preferred particle sizes for the four primary and overlapping disciplines are given in table I. Surface material composed of particulates up to 2 mm in diameter (fines) is considered the most likely to contain evidence of life and to provide straightforward evidence of surface alteration and volatiles. The size distributions and composition of these fines could also be expected to provide an understanding of sedimentary activity. Rock chips may reveal bulk compositions, rock and mineral identifications, conditions of formation, and age of major events. These will be vital in establishing the history of Mars.

Table I shows that the minimum scientifically acceptable sample mass for one series of tests is 30g. An additional 20g should be provided for followup tests to the initial series of tests and for investigations suggested later as new techniques are developed. This brings the minimum quantity to 50g. It is highly desirable, of course, in a unique mission of this type, to make available to the scientific community a sample which is larger than these minimum requirements. After a reasonably detailed and realistic spacecraft system design was defined in the course of the overall study, it was determined that the impact on design and total systems mass would be negligible if the sample mass were increased from 50g to 200g. Hence, the MSSR sample mass was set at 200g. This additional mass will also increase the probability of acquiring sufficient amounts of solid particles in each group of particle sizes.

TABLE I - SCIENCE SAMPLE REQUIREMENTS

Field of interest	Type of investigations	Sample requirement	
		Preferred particulate makeup	Minimum mass, g
Biology	Evidence of life	Fines (≤2-mm diameter)	10
Biochemistry	Biological-organic compound analysis	Fines (≤2-mm diameter)	5
Geochemistry	Low temperature phases/past surface conditions	Fines (≤2-mm diameter)	10
	Geochronology	3 to 5 rock chips	2.5
Petrology	Evidence of igneous activity	3 to 5 rock chips	2.5
Total mass			30g

TECHNIQUES AND CONDITIONS FOR SAMPLE ACQUISITION AND HANDLING

The sample should be obtained by scooping or some other mechanical technique, not by aspirating, and should not be transferred by aspirating. Aspiration may produce undesirable heating and an ill-defined undesirable separation by particle size. The sampling device should be capable of coping with the types of surfaces listed below:

Dune sand

Loess

Rock

Hard rock with 1-mm fines

Lag gravel (desert pavement)

Lunar (nominal)

Caliche

Vuggy irregular rock surface with 1-mm fines

A sampler that can obtain samples from a bare surface of hard rock would be desirable but poses mass and complexity problems. As an alternative, if the local surface is rock, small pieces of rock and small pockets or cracks containing fines should be located and these materials acquired. It should be possible to meet this requirement even without imagery from the lander, by use of trial and error, with such sensors as a mass-sensing or fill-indicating device in the sample processor and loader and position and force sensors on the sampling device. The sampling device should have two horizontal degrees of freedom to permit sampling attempts at various positions and to increase the chance that a sample will be obtained.

The sample should preferably be taken from a depth which is less than 5 cm but greater than a few millimeters. Materials of biological interest are most likely to be close to the surface, but a slight cover is desirable to reduce effects of spacecraft exhaust on the sample. Since a somewhat scaled-up version of the Viking motors and descent profile are used, the sample should be taken from a position at least 2.5 m and preferably greater than 3.0 m from the exhaust center line. The 2.5-m requirement arises from the need to obtain fines from surfaces consisting of rock covered with 1-mm layer of fines which might readily be blown away by the exhaust. With other soil models 50 cm would probably be adequate.

No specific requirements for accurate postflight knowledge of lander position have been established. Knowledge of the horizontal position to less than 100 kilometers appears to be adequate; knowledge of the vertical position, in terms of atmospheric pressure, to a millibar or so appears to be desirable.

The spacecraft should be so oriented that the area to be sampled is in sunlight during most of the day. This is desirable to heat the surface in order to drive off exhaust gases within the soil. Also, the area to be sampled should not be permanently shadowed by natural features, since living organisms are less likely to be found in such an area.

The sample should be taken and sealed during the coldest period of the night, to insure maximum content of absorbed and condensed atmospheric gases at the solid surfaces, and it should be sealed in ambient temperature. Sufficient time should elapse between landing and sampling to permit dissipation of spacecraft exhaust gases.

Recommended scientific and engineering measurements to be made while on the Mars surface are as follows:

(1) Temperature of the soil at the time and place of sampling

(2) Temperature of sampling and sample-processing equipment during sample handling

(3) Humidity and its variation throughout the day

(4) Wind velocity

(5) Light intensity during the day at the sample position (This measurement is intended to determine whether the sample is from a location shaded throughout the day by natural features or by the spacecraft)

(6) The local slope (lander attitude)

(7) Mechanical loads on the sampler

(8) Position of the sampler (elevation, azimuth, and boom extension)

(9) Atmospheric temperature

(10) Forces during touchdown (leg loads)

(11) Landing accelerations

(12) Atmospheric pressure

It is recognized that maintaining the sample temperature below a design goal level of -30° C may be difficult; but at no time should the sample temperature exceed 20° C for more than a few minutes. No special radiation shielding should be required to keep radiation exposure below a safe level. An extremely reliable vacuum seal is essential. A spring-loaded plug should be used to maintain a slight compressive load on the sample to prevent mechanical abrasion. Temperature, radiation, and pressure in the sample container should be monitored during the return trip to Earth.

CONTAMINATION BY PLANETARY ORGANISMS

Planetary quarantine considerations for MSSR missions apply to two planets; Mars and Earth. The twofold concern is to prevent contamination of Mars by terrestrial organisms and to prevent the contamination of Earth by possible Martian organisms (back contamination of Earth). This subject is treated briefly here, and more detail can be found in reference 1.

PROTECTION OF MARS FROM EARTH ORGANISMS

Current policies against contaminating Mars with Earth organisms will likely be in effect for an MSSR mission. These policies require that systems which land on Mars be sterilized and that precautions be taken to preclude accidental impact of nonsterile systems on Mars.

These requirements can be met, in general, by adopting the Viking sterilization techniques. The designs developed here took into account the effects of heat sterilization, and the systems will be encapsulated in a bioshield until after departure from Earth. A few types of components, such as the solid-propellant motors on the Mars ascent vehicle, are not related to Viking. Sterilization technology derived from ongoing development studies at JPL can be applied in these areas.

Corrections in the interplanetary trajectory are applied in a manner to avoid accidental impact on Mars in case of failure during Earth-to-Mars cruise, and modules which will be discarded after Mars arrival are deflected to preclude accidental impact. The orbiters are not sterilized for the MOR missions, and the Mars orbits for these missions were selected to guarantee long lifetimes.

PROTECTION OF EARTH FROM MARS ORGANISMS

Scientists are divided in their estimates on the possibility of Martian organisms and on what precautions should be taken to insure that Earth is not contaminated. There is also a large uncertainty about what insight will be gained from the results of the Viking '75 mission. Most current studies are aimed at determining sterilization techniques which will insure against Earth contamination but which will also permit maximum scientific value to be gained by analyses. This is a difficult challenge because the nature of Martian organisms, if they exist, is completely unknown.

The problems of controlling back contamination with a returned sample involve both interior contamination – contamination arising from the interior of the canister containing the sample, and from the sample itself – and exterior contamination – contamination arising from the exterior of the sample canister and from other spacecraft components which have contacted Martian surface and atmospheric material. An assessment of the problem of protecting Earth from Mars organisms on an MSSR mission is given in reference 1, and several design approaches are suggested.

One of the approaches was implemented during this study and is reported herein. It employs a bioshield for protecting the exterior of the Earth return vehicle. Details are discussed in the systems design section.

For all missions investigated, heliocentric trajectory corrections are applied to produce a maximum probability of accidental Earth impact of 10^{-8} in the event of spacecraft failure. Whereas the final Earth-to-Mars trajectory correction places the Mars entry capsule (MEC) on an entry trajectory, the final correction on the Mars-to-Earth trajectory is applied to preclude accidental impact.

MISSION MODE OPTIONS

The sketch of figure 1 illustrates the major phases of an MSSR trajectory and lists mode options for each phase. Modes for the six trajectory profiles selected for investi-

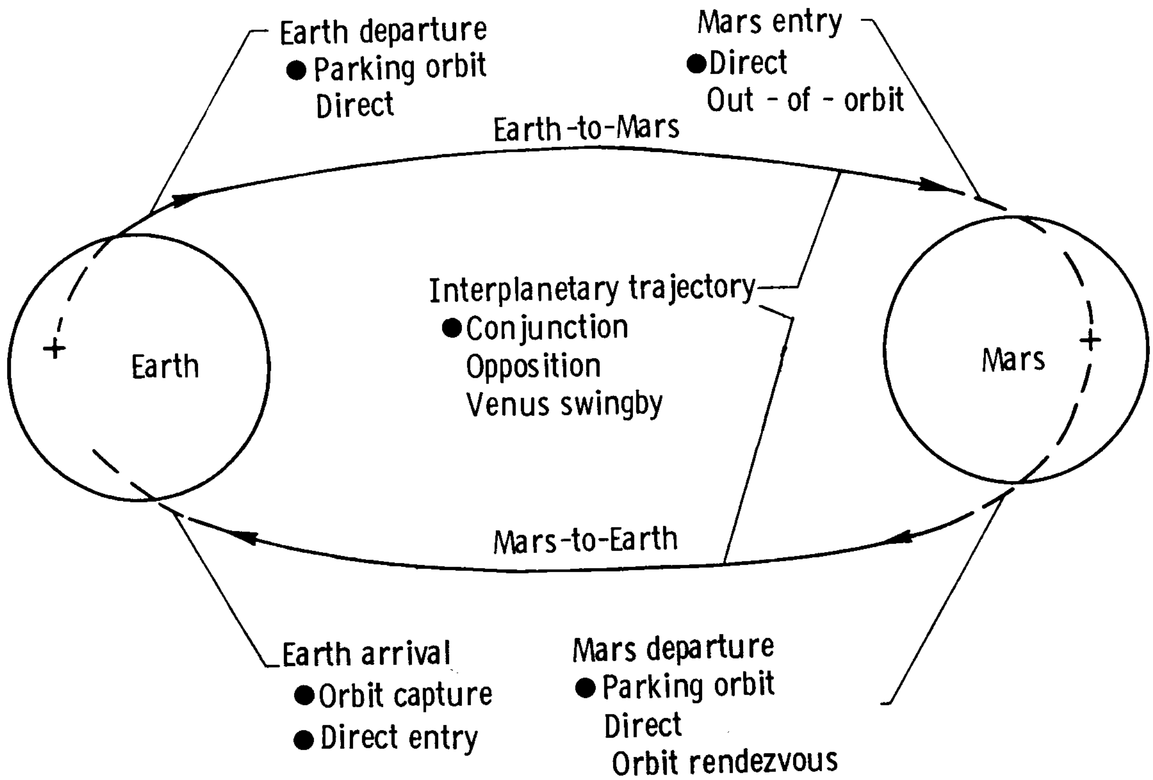


Figure 1.- MSSR trajectory phases and mode options. Solid dots indicate mode used for baseline mission.

TABLE II.- MISSION MODES FOR TRAJECTORY PROFILES

[Vehicles required for these modes are shown in table VI]

Trajectory profile		Trajectory phase				
		Earth departure	Interplanetary trajectory	Mars entry	Mars departure	Earth arrival
Baseline	1	Parking orbit	Conjunction	Direct	Parking orbit	Direct entry
Baseline	2	Parking orbit	Conjunction	Direct	Parking orbit	Orbit capture
Mars Orbit Rendezvous	3	Parking orbit	Conjunction	Direct	Rendezvous orbit	Direct entry
Mars Orbit Rendezvous	4	Parking orbit	Conjunction	Direct	Rendezvous orbit	Orbit capture
Mars Orbit Rendezvous	5	Parking orbit	Conjunction	Out of orbit	Rendezvous orbit	Direct entry
Mars Orbit Rendezvous	6	Parking orbit	Conjunction	Out of orbit	Rendezvous orbit	Orbit capture

gation in the study are given in table II. The profiles vary because of the different combinations of modes studied for the Mars entry, Mars departure, and Earth arrival phases. The two profiles in each of the 3 sets, (1-2), (3-4), (5-6), differ only in the alternate employment of the Earth entry and Earth orbit modes at Earth arrival. Modes for the baseline profiles, which were chosen on the basis of low energy requirements and relative simplicity of execution, are identified in figure 1. The feature which distinguishes the baseline profiles from the others is the employment of the parking orbit departure mode at Mars.

EARTH DEPARTURE

Departing Earth by means of a parking orbit has become fairly standard, and this mode can be used with either the Titan/Centaur or the Space Shuttle. This mode has the advantage of minimizing velocity penalties which result from off-nominal launch times, and it will be especially compatible with the Space Shuttle when that vehicle becomes operational.

INTERPLANETARY TRAJECTORY

The conjunction-class interplanetary trajectory was chosen over the other two options because of its lower overall velocity requirements as shown in figure 2. These data are representative for the 1979/1981 Earth launch opportunities. Mission time for the conjunction-class trajectory is longer, however, because of the year required at Mars. Spending an extended period in the uncertain Mars surface environment can be avoided by launching to a Mars orbit within a few days after landing; experience with Mariner and Pioneer has demonstrated that systems can operate reliably in a space environment for the periods required. Thus, sample-acquisition and related surface operations must be completed within a few days after landing, and thus all missions will employ an orbit-departure mode at Mars.

MARS ENTRY

The direct entry mode at Mars was selected for the baseline missions because it requires minimum energy and makes all the mass delivered to Mars, except that required to support the Earth-to-Mars cruise phase, available for entry. Direct entry is also employed on two of the profiles which employ the Mars orbit-rendezvous departure mode.

MARS ASCENT AND DEPARTURE

The overriding consideration of simplicity of execution led to the choice of the departure by means of a parking orbit mode for the baseline missions. A Mars orbit

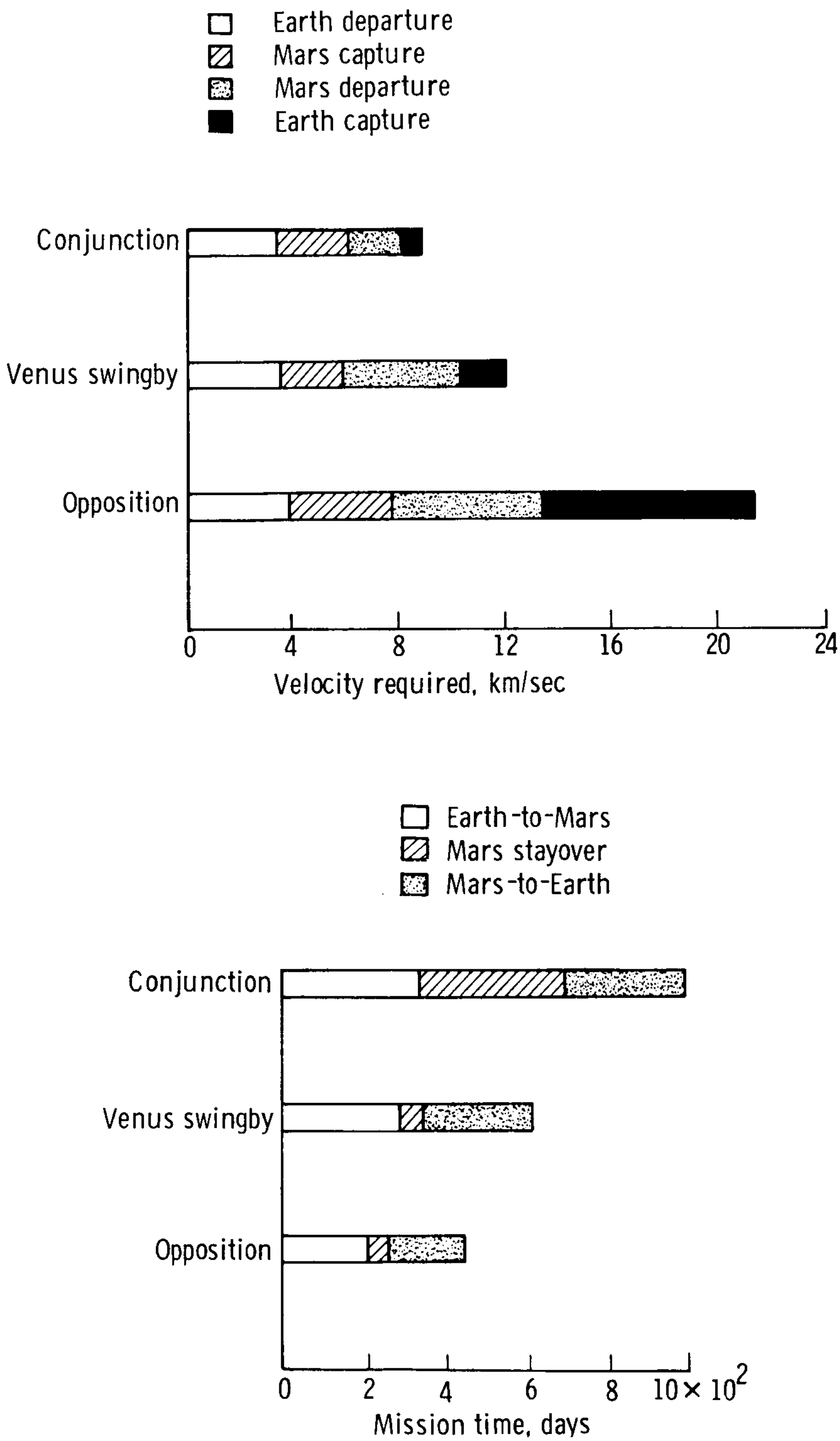


Figure 2.- Comparison of interplanetary trajectory classes.

rendezvous (MOR) mode, for example, requires two different vehicles at Mars, the Mars orbit vehicle (MOV) which will be inserted into orbit at arrival, and the Mars entry capsule (MEC). The MOR mode also requires an automated rendezvous and docking of the systems containing the sample with the MOV. It was anticipated that the choice of the Mars parking orbit departure mode would make greater demands on the entry and lander systems because these systems would be required to land the entire spacecraft for Earth return. It is believed that the advantage gained by a simpler implementation will outweigh this disadvantage.

EARTH ARRIVAL

The direct entry mode at Earth requires less energy and is relatively simple to execute. The orbit-capture mode offers more flexibility, however, in dealing with the back contamination problem. In-orbit sterilization and/or analyses may be possible with this mode or the sample could be retrieved and delivered to an Earth laboratory by utilizing the capabilities of the space shuttle and space tug. It was decided to give equal treatment to both the direct Earth entry and Earth orbit capture modes.

TRAJECTORY ANALYSIS AND DESIGN PARAMETERS

This section presents the following: (1) design parameters for the interplanetary trajectory, (2) Mars orbit analysis, and (3) results of the navigational error analysis.

INTERPLANETARY TRAJECTORY PARAMETERS FOR OPTIMUM LAUNCH OPPORTUNITIES

Table III(a) lists ranges of Earth launch and Mars arrival dates for optimum 30-day Earth launch periods during 1979 and 1981 mission opportunities and resulting trajectory parameters; table III(b) gives corresponding data for the Mars-to-Earth leg of the trajectory. Entry velocities at Earth and Mars are also shown in table III. These trajectories are essentially Hohmann transfers. Partial and total mission times are as follows:

Mission year	Time, days			
	Earth to Mars	At Mars	Mars to Earth	Total
1979	315	360	350	1025
1981	300	420	325	1045

TABLE III.- OPTIMUM LAUNCH PERIODS AND PARAMETERS FOR INTERPLANETARY TRAJECTORY

(a) Earth-to-Mars cruise, 30-day launch period

Parameter	1979 mission opportunity	1981 mission opportunity
Earth launch dates	10/14/79 to 11/12/79	11/09/81 to 12/08/81
Mars arrival dates	8/27/80 to 9/22/80	9/04/82 to 10/04/82
Geocentric injection energy, km^2/sec^2	12.2	12.2
Arrival hyperbolic excess velocity, km/sec	2.7	3.2
Declination of launch asymptote, deg	-36.0	-36.0
Mars entry velocity at altitude of 244 km, km/sec	5.5	5.8
Declination of Mars arrival asymptote, deg	-18.5	-26.0

(b) Mars-to-Earth cruise, 10-day launch period

Parameter	1979 mission opportunity	1981 mission opportunity
Mars launch dates	9/03/81 to 9/13/81	11/09/83 to 11/19/83
Earth arrival dates	8/19/82 to 8/30/82	9/28/84 to 10/05/84
Departure hyperbolic excess velocity, km/sec	2.5	2.3
Arrival hyperbolic excess velocity, km/sec	4.6	5.7
Declination of departure asymptote, deg	0	-12.0
Earth entry velocity at altitude of 122 km, km/sec	12.0	12.5

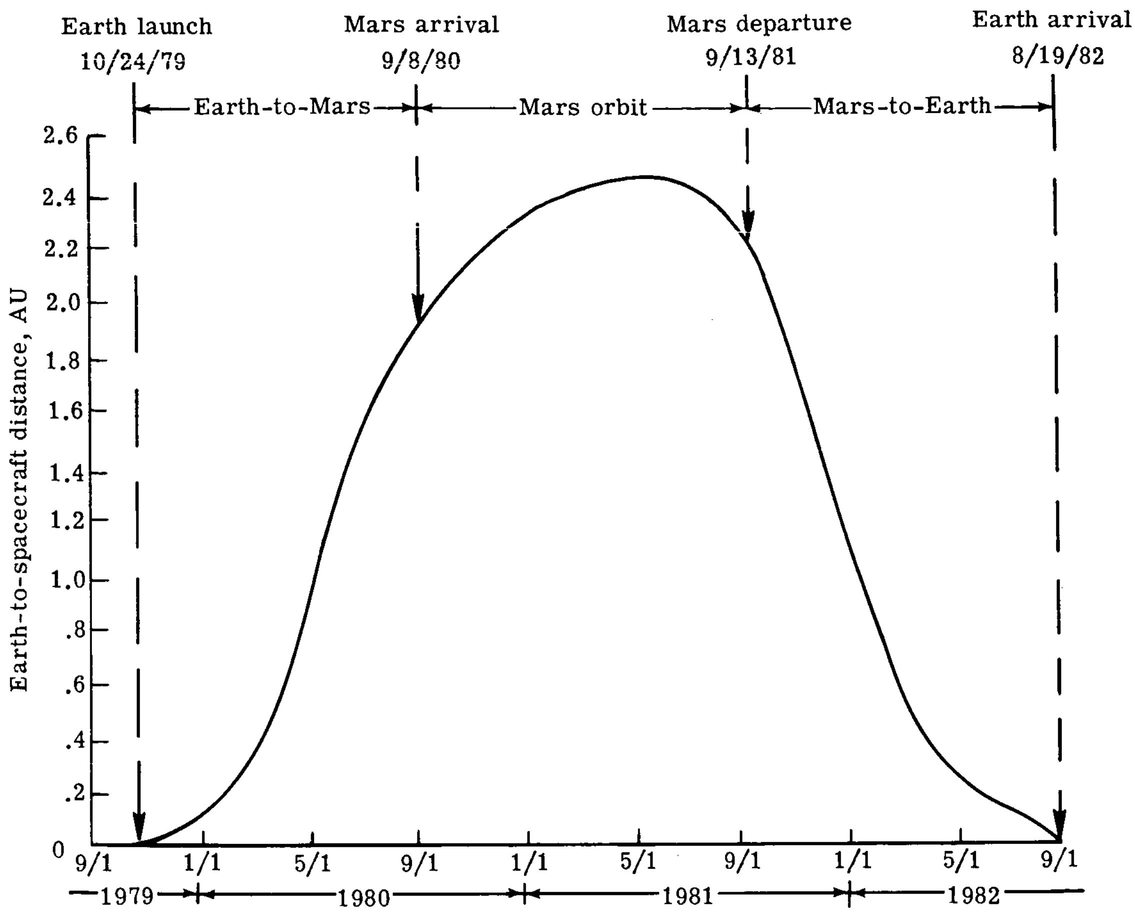


Figure 3.- Variation of Earth-to-spacecraft distance with mission time.

Figure 3 shows spacecraft-to-Earth distance as a function of mission time. During Mars orbiting, the communication distance to Earth reaches a maximum of about 2.5 AU. Figure 4 shows spacecraft-to-sun distance which is seen to vary between 1 and 1.6 AU.

MARS ORBIT ANALYSIS

The Mars ascent vehicle will insert into orbits of 35° inclination for both the baseline and Mars orbit rendezvous (MOR) missions. These orbits will permit a due-east launch from a latitude of 35°, the highest expected for the landing site. The orbits must satisfy orbital lifetime requirements and permit efficient insertion into Mars-to-Earth heliocentric trajectories. The latter requirement dictates orbits with (1) precession rates which will place the departure asymptote in the orbital plane at the desired departure date, and (2) line-of-apsides rotation rates which will permit insertion at the orbit periapsis. For the MOR missions, the orbit plane must also contain the departure asymptote at Mars arrival.

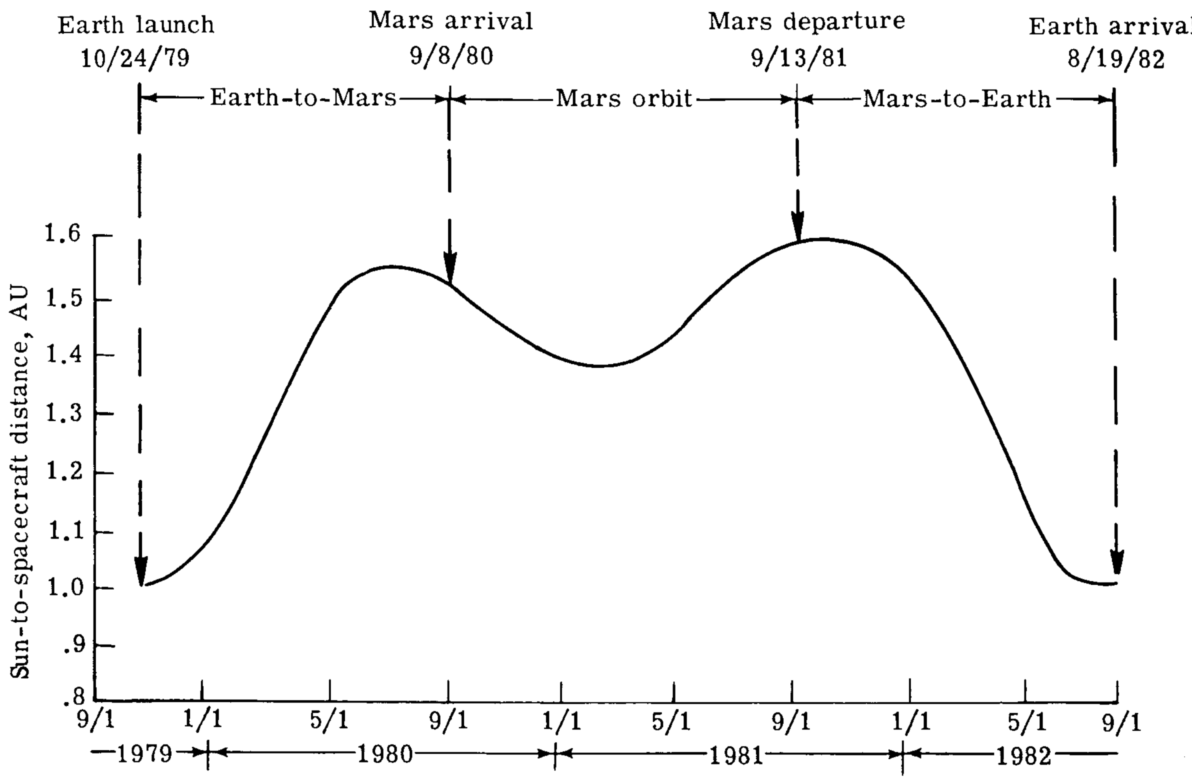


Figure 4.- Variation of Sun-to-spacecraft distance with mission time.

Baseline Missions

For the 1979 baseline missions the dwell time in Mars orbit is 365 days. Computations to determine 1-year lifetime orbits were made by using the Viking mean model atmosphere. Figure 5 shows a plot of this model to an altitude of 260 km. The combinations of periapsis and apoapsis altitudes required are shown in figure 6. For periapsis altitudes less than 325 km, eccentric orbits are required. For example, it can be seen that a periapsis altitude of 300 km requires an apoapsis altitude of 470 km. A much more conservative orbit than the 1-year lifetime orbit was found to be required for efficient orbit insertion. It has a periapsis altitude of 300 km and an apoapsis altitude of 4765 km. (See table IV.) Figure 7 shows that the velocity penalty due to injection delays from this orbit are very small compared with those for a 300-km circular orbit. For the 1981 baseline missions, dwell time in orbit is increased to 410 days. Allowing for the change in the declination of the departure asymptote (see table III) from 0° in 1979 to -12° in 1981, an orbit of 300 by 6145 km was found to meet injection requirements for the 1981 baseline missions. Mission design velocities for all the Mars orbits are given in table V.

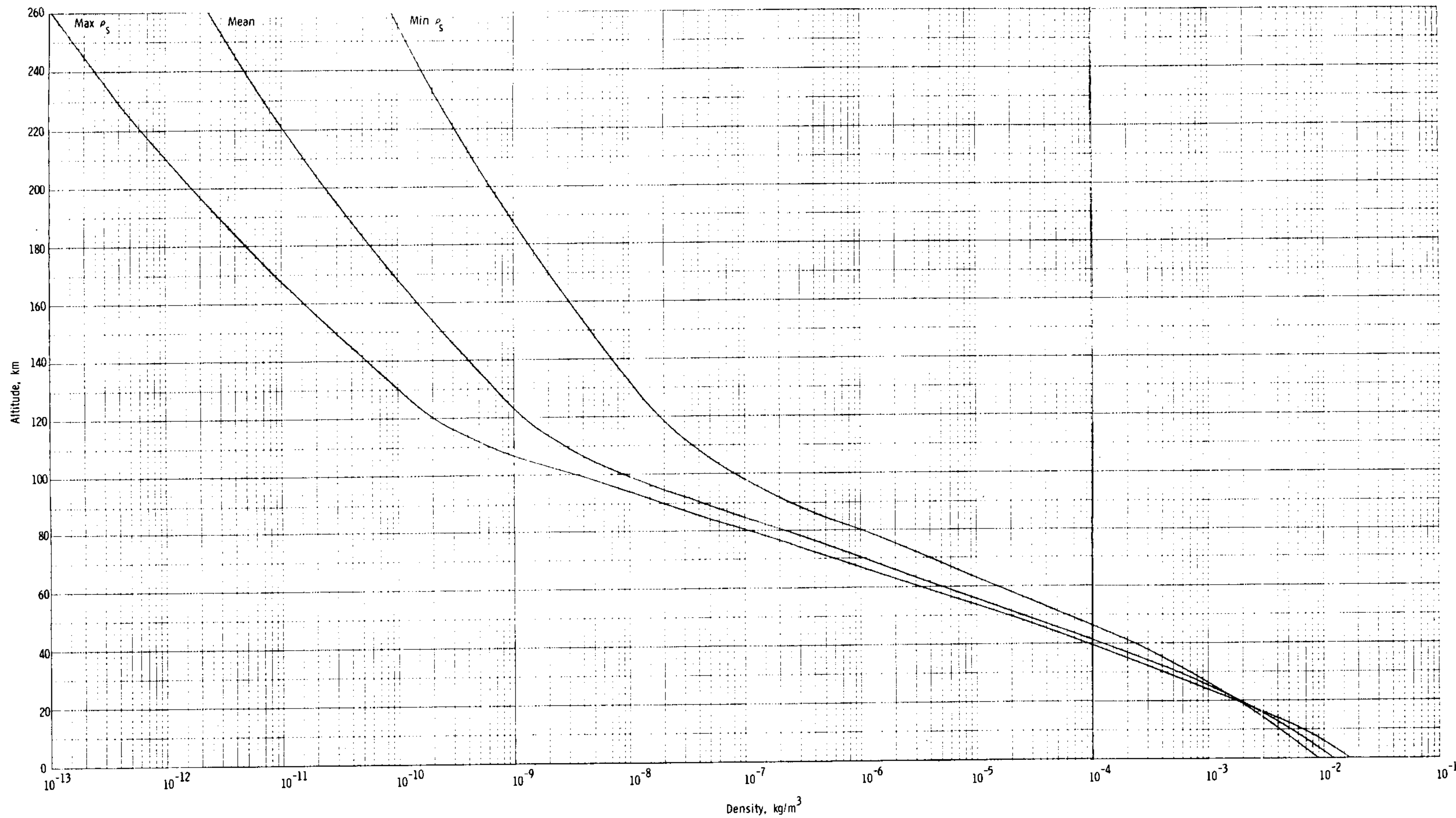


Figure 5.- Mars atmosphere density models.

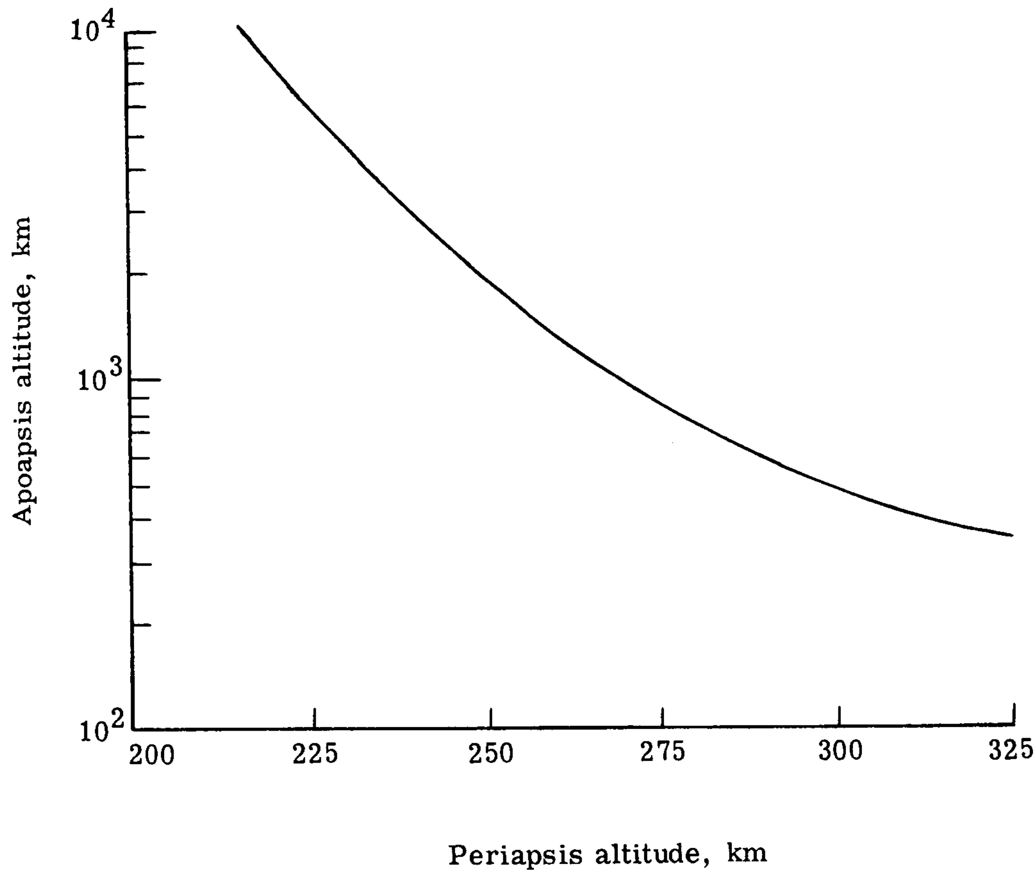


Figure 6.- Minimum apoapsis-periapsis requirements for 1-year-lifetime Mars orbits. Viking mean model atmosphere; inclination, $\leq 35^{\circ}$; ballistic coefficient, 110 kg/m^2.

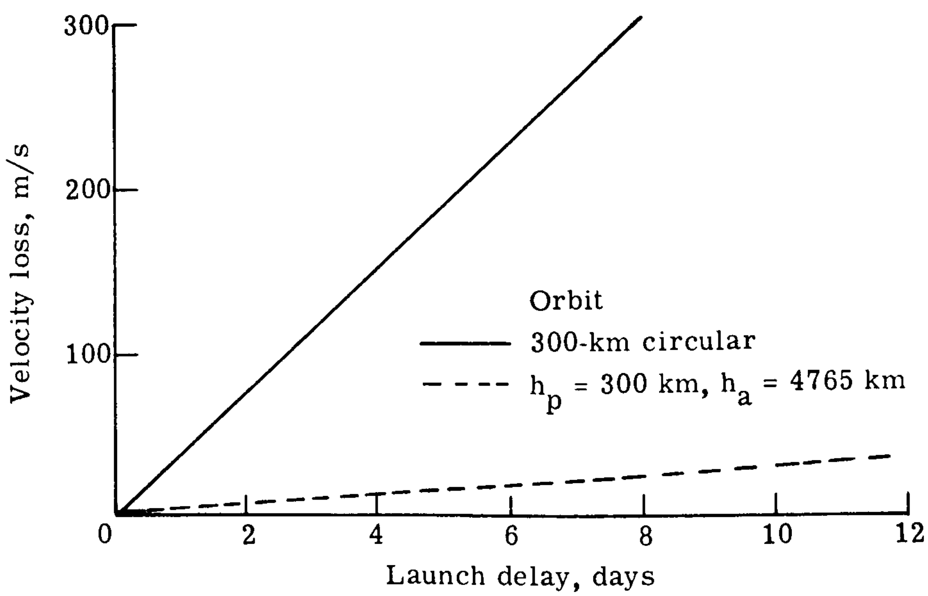


Figure 7.- Velocity penalty for injection delays.

TABLE IV.- CHARACTERISTICS OF MARS ORBITS

(a) Baseline mission

Parameter	1979 launch	1981 launch
Altitude of periapsis, km	300	300
Altitude of apoapsis, km	4765	6145
Minimum velocity to orbit, km/sec	4.28	4.37

(b) MOR missions

Parameter	1979 launch	1981 launch
Altitude of periapsis, km	872	915
Altitude of apoapsis, km	9800	7400
Minimum velocity to orbit, km/sec	4.65	4.56

TABLE V.- MARS ORBIT-RELATED DESIGN VELOCITIES

	Baseline missions		MOR missions	
	1979	1981	1979	1981
Velocity for insertion at Mars arrival, m/s			1331	1606
Velocity for surface-to-orbit insertion, m/s	4279	4373	4650	4557
Velocity to depart Mars orbit, m/s	1404	1275	1220	1273

MOR Missions

The MOVs will remain in Mars orbit after mission completion, and so the orbits for the MOR missions were chosen to survive till at least the year 2018 to meet quarantine standards set by COSPAR. For the earliest mission in 1979, a 38-year lifetime orbit is required. To be conservative, only orbits with at least 41-year lifetimes were considered. The required combination of periapsis and apoapsis altitudes are shown in figure 8. It was found that an 872-km by 9800-km orbit will precess to contain the departure asymptote. Heliocentric trajectory insertion will take place slightly off periapsis (about 4°) but this will produce only a slight velocity penalty. A 915-km by 7400-km orbit was found to meet all requirements for the 1981 missions.

Characteristics of the MOR orbits are given in table IV(b). Both orbits have apoapsis altitudes greater than those required for the 41-year lifetime.

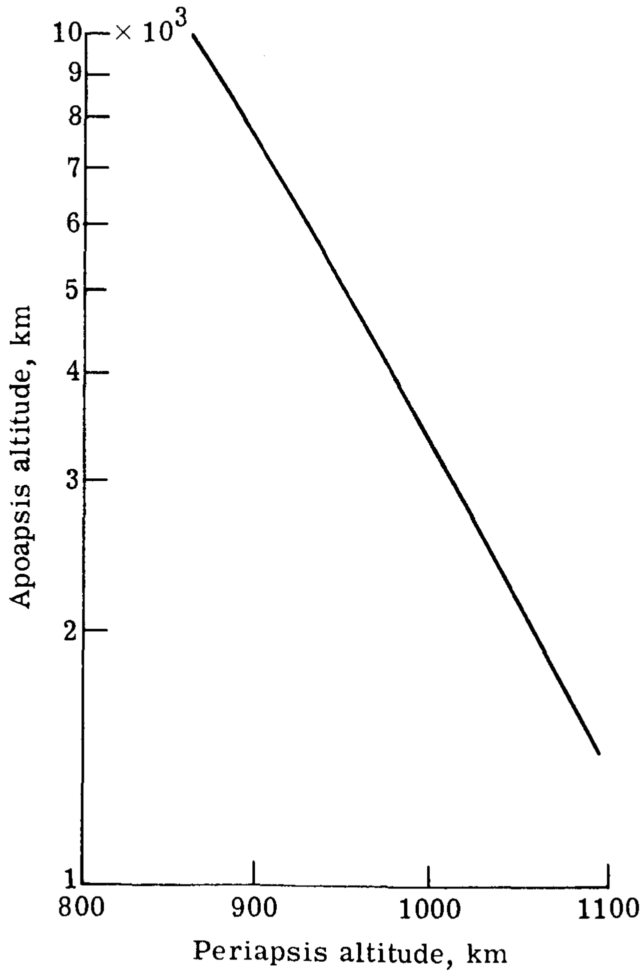


Figure 8.- Minimum apoapsis-periapsis requirements for 41-year-lifetime Mars orbits. Viking mean model atmosphere; inclination, $\leq 35^{\circ}$; ballistic coefficient, 110 kg/m^2.

NAVIGATIONAL ERROR ANALYSIS

An analysis was performed to determine velocities required to correct errors in the interplanetary trajectory and the Mars orbits and to determine entry angle corridors at Mars and Earth. Below are listed velocities which were found to be necessary to correct 99 percent of the errors in the interplanetary trajectory and the Mars orbits.

Source	Velocity, m/s	Condition
Earth-Mars interplanetary phase	20	$P_{IQ} = 10^{-6}$
Orbit trims to correct for stages 1 and 2 launch errors and station keeping	100	100-percent correlation between stage errors
Mars-Earth interplanetary phase	145	$P_{IQ} = 10^{-8}$

Factors In Determining Correction Velocities

Earth-to-Mars cruise.- Factors considered include Earth launch errors, uncertainties in the determination of the interplanetary trajectory, errors introduced during the performance of midcourse maneuvers, and quarantine biasing errors. Four midcourse maneuvers are assumed. The first three are biased to prevent accidental impact of the spacecraft on Mars. The maximum probability that impact would occur in the absence of a final maneuver is denoted by P_{IQ}. For the direct entry mode on both the baseline and MOR missions the final maneuver places the spacecraft on the desired entry trajectory. For these missions the supporting module must be deflected after separation to prevent its accidental impact on Mars. For the out-of-orbit entry mode, the final maneuver places the spacecraft on a favorable trajectory for subsequent orbit insertion.

Mars-to-Earth cruise.- Essentially the same factors are considered as for the Earth-to-Mars cruise phase. Note that the value of P_{IQ} is two orders lower than that for Mars arrival. The final maneuver for both the Earth entry and the orbit capture modes are designed to maintain the spacecraft on the nonimpact trajectory $\left(P_{IQ} = 10^{-8}\right)$, and hence the Earth entry capsule must be deflected after separation to place it on the desired entry trajectory.

Mars orbit errors and station keeping.- Velocity to correct errors during Mars ascent was obtained for the baseline mission only (300-km by 4765-km orbit). This velocity will correct for errors introduced during the surface-to-orbit phase and allows for station keeping orbit-trim maneuvers. A direct correlation was assumed between errors produced by the two stages of the Mars ascent vehicle.

Entry Angle Corridors

Ninety-nine percent entry angle corridors for arrival at Mars and Earth are given in the table. Final orbit determination and trajectory corrections are assumed to be made

	$\Delta\gamma_E$, deg	γ_E, deg
Mars	±1.00	≦-20.0
Earth	±0.17	≦-14.0

no earlier than E - 6 hours. The values for Mars arrival apply for the direct entry mode for both the baseline and MOR missions. The 2° entry corridor at Mars, which is half that for Viking, results from improved targeting capability which will be available by 1979.

GENERAL SYSTEMS DESIGN AND INTEGRATION

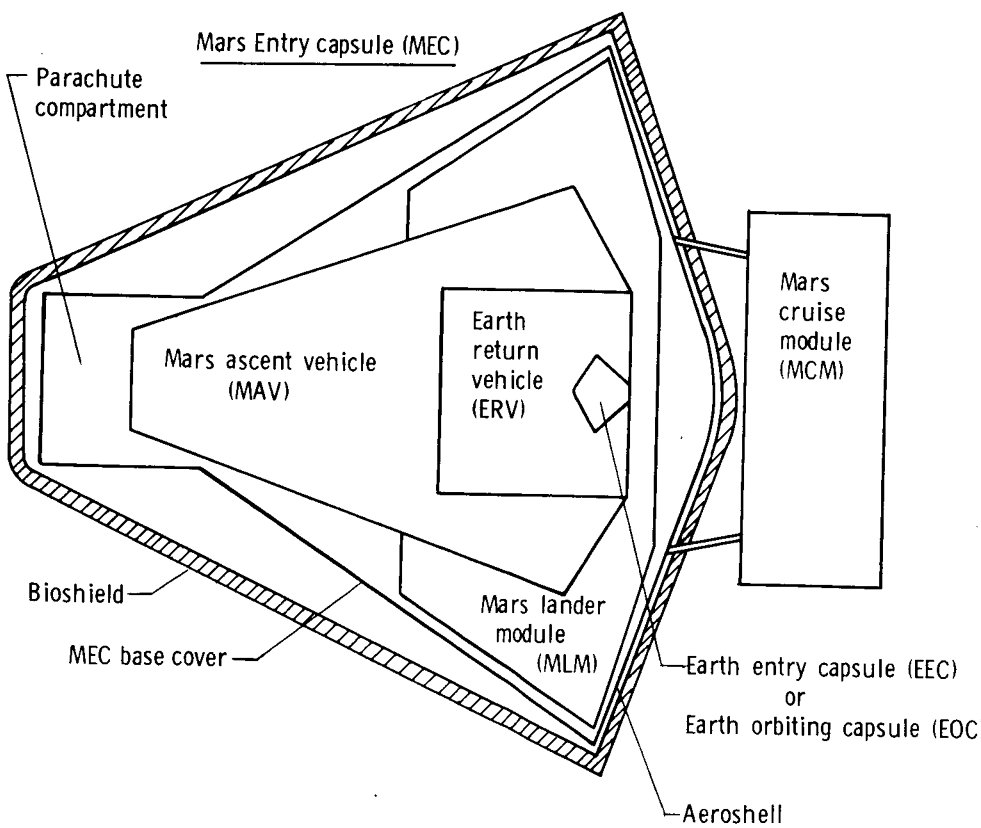
The principal rationale for all systems design was to develop integrated systems which will efficiently and reliably return an adequate sample to Earth. Functional requirements of the systems to implement each mode were considered. Technology developed in other planetary programs was emphasized, and compatibility of systems was sought to minimize integration problems and duplication of functions. Where practical, systems were selected for compatibility between the baseline and Mars orbit rendezvous (MOR) missions. This section describes the general evaluation of systems requirements, the rationale for selecting classes of systems, and the integration considerations. Analyses and more detailed design considerations are covered in later sections.

SPACECRAFT AND MODULAR COMPONENTS

The MSSR spacecraft consists of two major bodies: (1) a Mars entry capsule (MEC) patterned after the Viking entry body, and (2) either a Mars cruise module (MCM) or a Mars orbit vehicle (MOV). Sketches of the spacecraft showing module locations are given in figure 9. The MEC is encapsulated in a Viking-type bioshield until after Earth departure. The aeroshell, parachute systems, and Mars lander module (MLM) provide a 3-stage deceleration sequence like that of Viking. Entry trajectory and entry and landing systems analyses were performed to determine modifications to Viking systems required to support the more demanding entry of the MSSR missions.

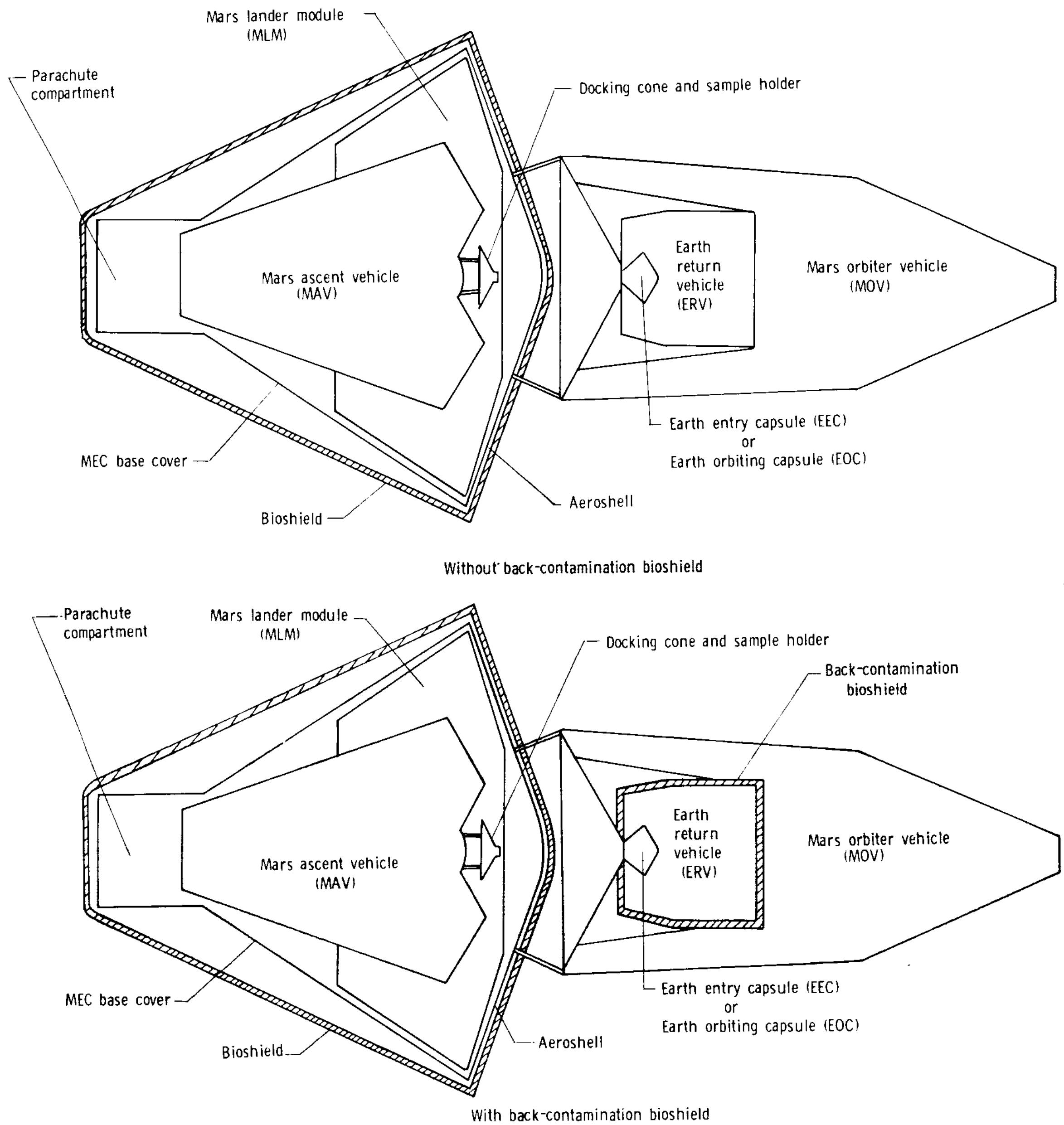
During Earth-to-Mars cruise, essentially all functions are performed by the Mars cruise module (MCM) on the baseline missions and by the Mars orbit vehicle (MOV) on the MOR missions. The MCM and MOV design concepts are based on the Viking orbiter design with systems scaled down to reflect the absence of a science payload and much-reduced data handling requirements. The MOV will insert into orbit at Mars arrival where it will later rendezvous with the Mars ascent vehicle for sample transfer to the Earth return vehicle (ERV). It will then separate from the ERV and will have no further functions. The MCM is substantially smaller than the MOV because the former does not insert into Mars orbit.

In designing components for continuing the mission from Mars launch to Earth arrival, the approach was to start at the Earth arrival phase and work backward. The Earth entry capsule (EEC), which is designed to propulsively deflect itself for a direct atmospheric entry at Earth, is based on the technology of reference 2, and the Earth orbit capsule (EOC), which is designed to propulsively brake itself into Earth orbit, is based on the same technology. The ERV, which serves as an orbiter during most of the year at Mars, provides propulsion for Mars departure and transports the EEC (or EOC) back to Earth, is based on Pioneer technology. The ERV systems designs are essentially the same for baseline and MOR missions, but requirements are more demanding on baseline missions because the ERV performs important functions during Mars descent and ascent. It is a quasi-cylindrical configuration utilizing spin stabilization. The Mars



(a) Baseline.

Figure 9.- Modular sketches of MSSR spacecraft.



(b) Mars orbit rendezvous.

Figure 9.- Concluded.

ascent vehicle (MAV) launches the ERV and EEC (or EOC) from the Mars surface into orbit for baseline missions, and it launches the sample canister and systems for rendezvous and sample transfer into orbit for MOR missions. Propulsion parameter and trajectory optimization studies were performed to support MAV systems designs.

The modular components are all designed for specialized roles during some part of the mission, but they are also highly integrated to complement each other. The block

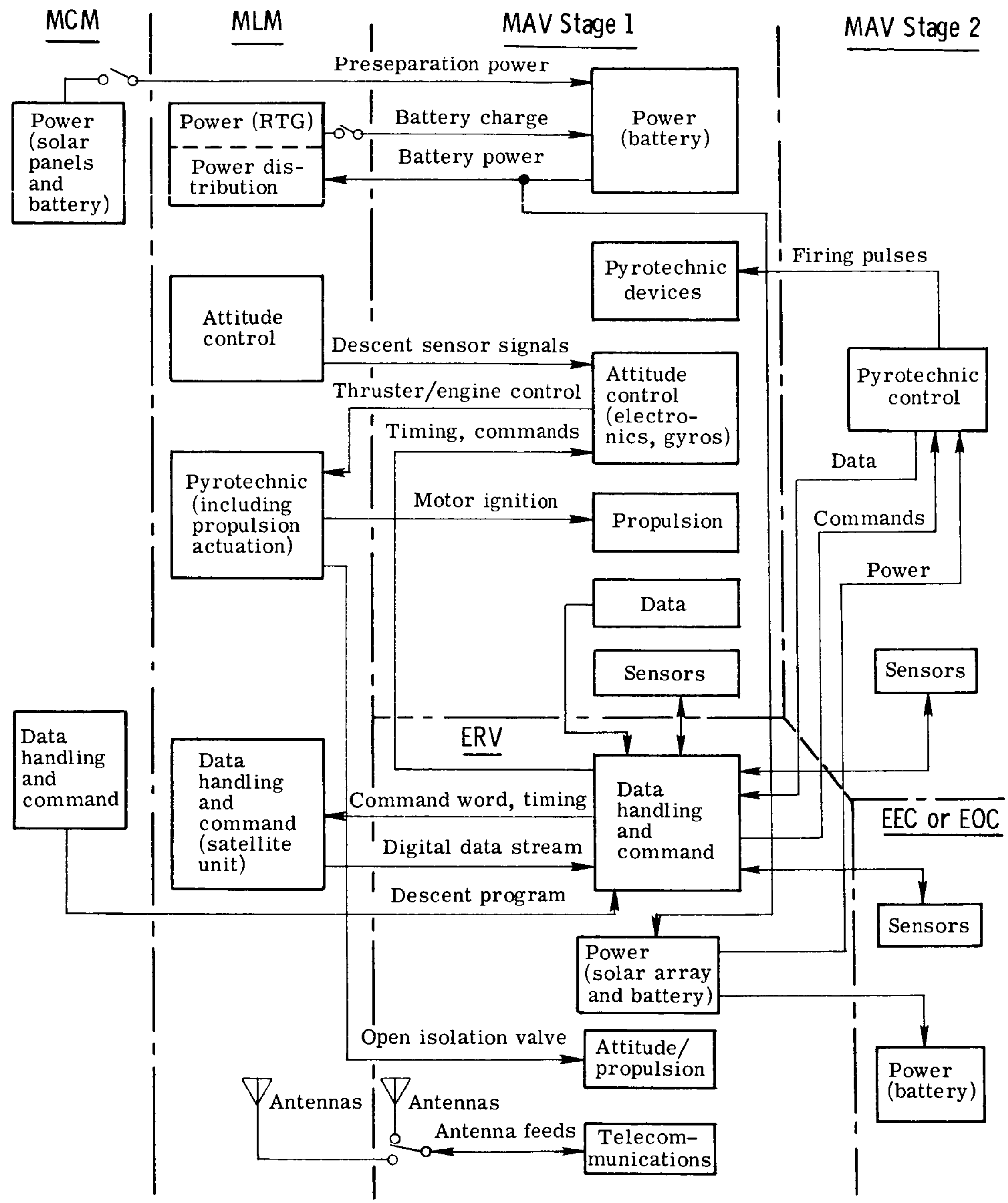


Figure 10.- MSSR spacecraft module interfaces for baseline mission.

diagram in figure 10 illustrates the modular interfaces of the integrated spacecraft for the baseline missions.

Emphasis in the current study was on developing systems for missions which could be performed by utilizing a single Titan III E/Centaur as a launch vehicle, and so payload

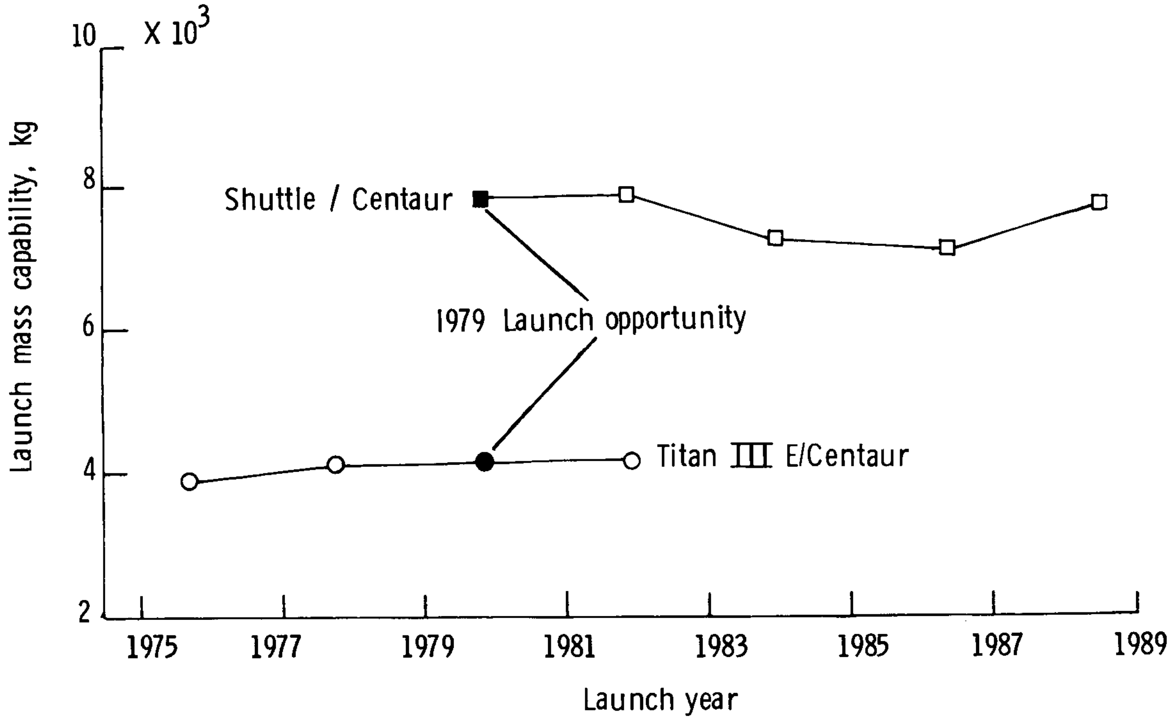
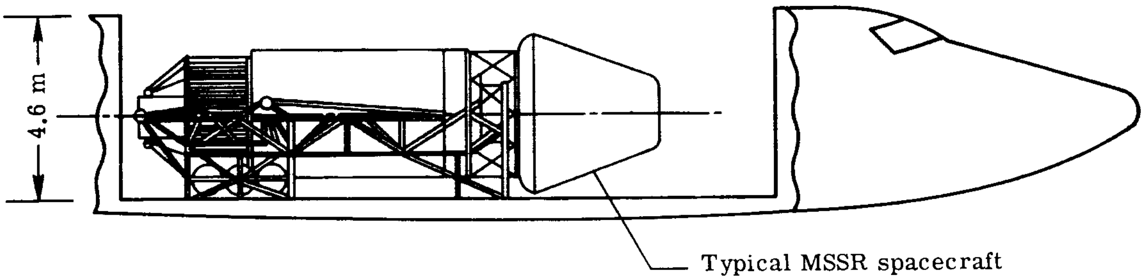
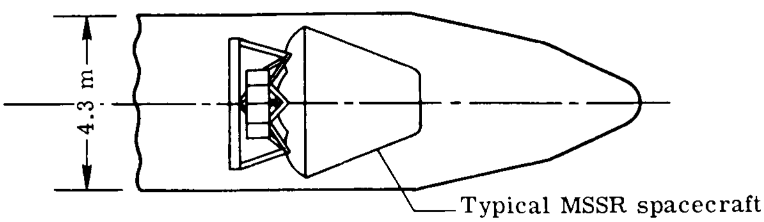


Figure 11.- Vehicle payload capability for Earth-to-Mars launch.



Space shuttle



Titan III E/Centaur

Figure 12.- Payload compartments of space shuttle and Titan III E/Centaur.

mass and size constraints of that vehicle were imposed. Figure 11 compares payload mass capability of the Titan III E/Centaur with that of the Space Shuttle/Centaur, and figure 12 shows sketches of the payload compartments of the Titan III E/Centaur and Space Shuttle. Of the two constraints, the payload size constraint is the more severe because it limits the maximum allowable diameter of the Mars entry aeroshell. In fact, it will be seen that this is the main constraint on the baseline missions because it limits the mass landed on Mars which will be available for launch from Mars.

PROPULSION AND ATTITUDE CONTROL

Propulsion and attitude control required during interplanetary cruise and at Mars arrival will be located on the MCM for baseline missions and on the MOV for MOR missions. Essentially, one system performs both propulsive velocity changes and attitude control. Velocity for trajectory corrections is applied through a two-axis gimbaled motor, and thrusters are provided for attitude control. The system is based on Mariner 9 design, except that the attitude-control system utilizes a high efficiency hydrazine monopropellant, instead of the cold gas employed on Mariner. After separation from the MEC, the MCM must propulsively deflect itself to prevent Mars impact. The MOV propulsion system must insert the MOV into orbit if direct entry is employed and must insert the MOV and the MEC if out-of-orbit entry is employed. The MOV attitude propulsion system must also provide attitude control during phase-one orbiting, assist in rendezvous and docking, and orient the MOV for release of the ERV. Phase-one orbiting is that period from orbit insertion at Mars arrival to ERV separation, and phase-two orbiting is that period from ERV separation to Mars departure. The propellant and tankage requirements are substantially greater on the MOV than on the MCM.

The EEC and the EOC each have three small solid-propellant rocket motors. The motors on the EEC deflect it onto an entry trajectory, and the motors on the EOC provide velocity for braking into an Earth orbit. Both types of capsules will be spinning when they separate from the ERV and have no active attitude-control systems.

The ERV will have a Pioneer 9-type attitude/propulsion system which will be used for orbit trim maneuvers and attitude control during orbiting and which will maintain attitude control and perform trajectory corrections during Mars-to-Earth cruise. Additionally, a solid-propellant rocket motor on the ERV will provide velocity for Mars orbit departure. Two-stage, solid-propellant Mars ascent vehicles were selected for all missions on the basis of vehicle propulsion and trajectory optimization studies. The ascent vehicles for the baseline missions will have three-axis controlled first stages and spin-stabilized second stages. Some MOR missions employ vehicles with spin-stabilized second stages, and some employ vehicles with three-axis controlled second stages.

TELECOMMUNICATIONS

A plot of telecommunications range during the mission is shown in figure 3. From this range requirement and the fact that no science instruments will be flown which require high data-rate transmission, relatively simple equipment, based entirely on current technology, will be used for all telecommunication requirements. A Pioneer 9 type radio located on the Earth return vehicle (ERV) will provide communications during all phases of the baseline mission except the Earth-to-Mars cruise phase when ranging capability is required. A Mariner 5 type radio is located on the MCM to satisfy this requirement. A Pioneer 9 radio on the ERV will provide telecommunications during phase-two orbiting and Mars-to-Earth cruise for MOR missions, and a Mariner 5 radio with ranging capability will be located on the MOV for communication during Earth-to-Mars cruise and phase-one orbiting.

For MOR missions which employ Mars ascent vehicles with three-axis controlled second stages, a telecommunications system identical to that on the ERV will be located on the MAV second stage. It will provide a two-way communication link with Earth while on the Mars surface and will be used during the rendezvous and docking operations. For MOR missions which employ spin stabilization on the second stage of the MAV, a telecommunications system identical to the one on the ERV will be located on the MLM for transmission during surface operations, and a system on the second stage of the MAV will be used during rendezvous operations. The relatively short periods during descent and ascent at Mars make communications with Earth during these periods virtually impossible. The descent and ascent programs are transmitted from Earth prior to these periods and are stored in the data handling and command systems.

DATA HANDLING AND COMMAND

A review of existing flight data handling and command (DHAC) systems such as those used on Mariner and Pioneer revealed that current technology would provide capability in excess of that required for MSSR missions. It was also concluded that current systems were too large and massive. A preliminary design of a new DHAC system was, therefore, conceived that would make use of current technology, which meets all the requirements of the MSSR missions but which is substantially smaller and less massive than current flight systems. Pertinent design features of the system are described in the section on the Earth-return vehicle. One of these DHAC systems is located on the ERV for all missions. These DHAC systems perform most of the data handling and command functions during the baseline missions except that an identical DHAC system on the MCM performs the required functions during Earth-to-Mars cruise.

One of the new DHAC systems is located on the MOV, and it provides capability during Earth-to-Mars cruise and phase-one orbiting. One of these systems is also located on the second stage of the MAV for all MOR missions and provides data handling and command capability during Mars descent, surface operations, Mars ascent, and rendezvous and docking. A partial DHAC system is located on the MLM for all missions to provide the additional capability required during surface operations.

POWER

Analysis showed that existing technology could be used to supply power for MSSR missions. A scaled-down Mariner 9 power system will be located on the MCM (baseline missions) and the MOV (MOR missions) to provide power during Earth-to-Mars cruise. The system on the MOV will also provide power during phase-one orbiting. During rendezvous, the solar panels on the MOV will be articulated. Power for entry on both types of missions will be supplied by a Viking-type radioisotope thermoelectric generator (RTG). A battery on the MAV will provide power needs in excess of RTG output during surface operations and will take care of transients.

The power required during thrusting of the first stage of the MAV will be supplied by the battery on the MAV. It will be located on stage one for baseline missions and for MOR missions employing spin-stabilized second stages. It is located on the second stage of the MAV which employs three-axis controlled second stages. The second stage of the MAV for the baseline missions requires only pyrotechnics power which is supplied by the ERV. The Mars ascent vehicles for MOR missions have power sources on their second stages to provide power during rendezvous and docking operations.

The ERV will be equipped with a battery which will be charged by an array of solar cells attached to the external surface of the ERV. Pioneer 9 type power distribution is employed, and to conserve power the main power user, the ratio transmitter, is turned on by ground command intermittently. The EEC and EOC are each equipped with a small battery which will be trickle-charged until separation from the ERV.

STRUCTURE, THERMAL DESIGN, MECHANICAL DEVICES, AND PYROTECHNICS

The structural designs of the MCM and MOV are based on Mariner technology as is the Viking orbiter. The MEC is derived from Viking technology. The MLM structure is partly based on Viking technology, but some new design was required to minimize mass and to accommodate the MAV. The ERV structural design was based on Pioneer 9 technology. The EEC and EOC structural designs were based on small-satellites technology because of severe constraints on mass and volume.

No new technology was required for thermal control. Designs are based on Mariner, Viking, and Pioneer technology.

The designs of essentially all the pyrotechnic and mechanical devices including the practice of using dual-bridge wire squibs for redundancy are based on Mariner technology.

DESIGN TO PREVENT EARTH CONTAMINATION BY MARS ORGANISMS

Of the several design approaches discussed in reference 1, one was selected for design implementation. The approach, which is applied to an MOR mission, uses a bio-shield to quarantine the ERV and EEC or EOC from Earth launch to separation of the ERV and MOV. (See fig. 9(b).) After sample canister transfer from the MAV, a combustible plug seals the opening in the EEC or EOC and heat-sterilizes the exposed surface of the canister.

SUMMARY OF MISSIONS INVESTIGATED

Systems concepts were derived for twelve missions during the course of this study. Table VI lists the variants of the major spacecraft components which support each trajectory phase for the twelve missions. Some component variations result primarily from

TABLE VI.- VARIATIONS IN TRAJECTORY PROFILES AND MAJOR SPACECRAFT COMPONENTS FOR MISSIONS INVESTIGATED

[MOVs support the Earth-to-Mars phase and phase-one orbiting.
ERVs support the Mars-to-Earth phase and phase-two orbiting.
Phase-one orbiting: orbit insertion at Mars arrival (MOR missions only) through ERV/MOV separation.
Phase-two orbiting: period during which ERV serves as orbiter (after ERV/MOV separation on MOR missions).]

Mission	Trajectory profile (table II)	Systems used for trajectory phases								
		Earth departure	Earth to Mars	Mars entry and landing			Mars ascent	Mars departure, Earth return	Earth arrival	
				Aeroshell	Parachute	Lander			EEC	EOC
1	1 (Baseline)	(B/S)$_1$	(MCM)$_1$	(A/S)$_1$	(P/S)$_1$	(MLM)$_1$	(MAV)$_1$	(ERV)$_1$	(EEC)$_1$	
2	2 (Baseline)		(MCM)$_1$	(A/S)$_1$		(MLM)$_1$	(MAV)$_1$	(ERV)$_1$		(EOC)$_1$
3	3 (MOR)		(MOV)$_1$	(A/S)$_2$		(MLM)$_2$	(MAV)$_2$	(ERV)$_2$	(EEC)$_1$	
4	4		(MOV)$_1$					(ERV)$_2$		(EOC)$_1$
5	3		(MOV)$_2$					(ERV)$_3$	(EEC)$_1$	
6	4		(MOV)$_2$	↓		↓		(ERV)$_3$		(EOC)$_1$
7	5		(MOV)$_4$	(A/S)$_4$		(MLM)$_4$		(ERV)$_2$	(EEC)$_1$	
8	6		(MOV)$_4$	(A/S)$_4$	↓	(MLM)$_4$	↓			(EOC)$_1$
9	3		(MOV)$_3$	(A/S)$_3$	(P/S)$_2$	(MLM)$_3$	(MAV)$_3$		(EEC)$_1$	
10	4		(MOV)$_3$	(A/S)$_3$		(MLM)$_3$				(EOC)$_1$
11	5		(MOV)$_5$	(A/S)$_5$		(MLM)$_5$			(EEC)$_1$	
12	6 ↓	↓	(MOV)$_5$	(A/S)$_5$	↓	(MLM)$_5$	↓	↓		(EOC)$_1$
Total variants . . .		1	6	5	2	5	3	3	2	

differences in mission modes employed. For example, the component which supports the Earth-to-Mars phase on the Mars orbit rendezvous (MOR) missions must also serve as a Mars orbiter. Other component variations result from changes in the system techniques for performing a given mission mode. For example, employing two types of stability control on the second stages of the Mars ascent vehicles (MAVs) will be seen to produce variations in several other major components such as the Mars orbit vehicles (MOVs) and the Mars lander modules (MLMs). Descriptions of systems follow, and summaries of systems masses for each component variant given in table VI are presented in their appropriate sections. Detailed mass breakdowns are given in appendix A. A sequence of major mission events is given in appendix B.

SYSTEMS DESIGN AND OPERATION FOR EARTH-TO-MARS CRUISE, MARS ARRIVAL, AND PHASE-ONE ORBITING

During Earth-to-Mars cruise the Mars entry capsule (MEC) is supported by the Mars cruise module (MCM) for the baseline missions and by the Mars orbit vehicle (MOV) for the Mars orbit rendezvous (MOR) missions. (See table VI.) The front part of the bioshield is separated from the MEC after Earth departure. For baseline missions the aft section of the bioshield is separated along with the MCM at Mars arrival and remains with the MCM on the flyby trajectory. For MOR missions the aft section is separated along with the MOV and is then separated from the MOV to prevent interference with the MOV operations.

The MCM performs supporting functions for the MEC during Earth-to-Mars cruise, and it performs a deflection after it separates from the MEC which prevents its accidental impact on the surface of Mars. The MOV performs essentially the same functions during Earth-to-Mars cruise on the MOR missions. In addition, it must provide velocity for orbit insertion and perform certain functions during phase-one orbiting (insertion to separation of the Earth return vehicle) which include assisting in rendezvous and docking and transferring of the sample canister from the Mars ascent vehicle (MAV) to the Earth return vehicle (ERV).

DESIGN AND OPERATION OF MARS CRUISE MODULE SYSTEMS

During Earth-to-Mars cruise, the MCM systems will make trajectory changes, maintain attitude, maintain two-way communication with Earth, and implement ground commands. Its data handling and command system receives and internally transmits the Mars descent program. A summary of MCM systems and their masses is given in table VII.

TABLE VII.- SUMMARY OF SYSTEM MASSES FOR

MARS CRUISE MODULE

System	Mass, kg
Structure	66.5
Telecommunications	15.8
Power	25.5
Data handling and command	4.5
Pyrotechnics	5.4
Cabling	23.6
Temperature control	7.9
Mechanical devices	7.5
Propulsion and attitude control	70.0
Total	226.7

Propulsion and Attitude Control

A single system is used for making velocity changes and maintaining attitude control. Except for the use of hydrazine propellant, the system is essentially the same as that on the Mariner 9 and the Viking orbiters. Twelve thrusters in a redundant system are used to control attitude in three planes, and one gimbaled motor provides velocity for trajectory changes. Inertial reference is provided by a system employing roll, pitch, and yaw gyros and an accelerometer, and celestial reference is provided by Sun sensors and a Canopus tracker. These attitude reference units are also used in conjunction with the gimbaled motor for thrust-vector control. The propellant mass allocated includes that required to maintain attitude control and to provide velocities for the 20 m/s midcourse trajectory correction and 60 m/s for the deflection maneuver after separation of the MEC.

Structure

The MCM structure system consists of the bus (Mariner design inheritance), with its electronic bays, and structural supports for propellant tanks, antennas, solar panels, bioshield, and the MEC.

Telecommunications, Data Handling, and Command

The telecommunications system on the MCM provides for receiving commands by a low-gain antenna and transmitting telemetry data at 8 or 16 bits per second by a low-gain antenna (near Earth) or a high-gain antenna from Earth departure until after separation of the MEC. The system is comprised of a 10-watt S-band radio with ranging provisions (Mariner 5 design inheritance), and two antennas (high gain and low gain) with their respective coaxial cables and feeds. This radio includes redundant receivers, transmitters, and command decoders. The design of the MCM high-gain antenna is inherited from the Mariner 9 design. It is a 23-cm-diameter horn with a length of 53 cm. Its characteristics include a peak gain of 15 dB and a full beam width of 36°. After its initial deployment this antenna remains fixed.

The low-gain antenna is the biconical type, with the lobes of its gain pattern tilted by 20°. The complete pattern is a toroid of revolution, and the axis of symmetry allows for large variations of the Earth clock angle near Earth. The peak gain of this antenna is 2 dB and the full meridian beam width is 70°.

The MCM data handling and command system is identical to that used on the ERV and is described in the section on the ERV. An electrical interface between the data handling and command systems in the MCM and ERV permits the descent program to be sent to the data handling and command system in the ERV where it is stored in a solid-state memory until required for controlling the descent of the MEC.

Power

The MCM power system uses solar panels as power source and controls and converts this power to the ac and dc voltages required by the various MCM systems.

The four solar panels employ silicon photovoltaic cells and are deployable since they must be in a stowed position before and during launch. The battery was selected to be identical to that used on the MAV to reduce program cost and complexity. It is a silver-zinc battery with a 280 W-hr rating. Redundant boost regulators feed two redundant 2.4-kHz inverters and one 400-Hz inverter. The inverters are required by the attitude-control subsystem. A 30-V dc regulator provides power for the motor gimbal actuators. Two high-voltage converters provide the power required by the redundant traveling-wave tube power amplifiers in the radio transmitter. Redundant dc/ac/dc converters provide regulated voltages to other subsystems as required.

The battery is normally charged from the primary bus in flight; however, it can also be charged from external power during system test and prelaunch operations. The primary bus can also be switched to external power during these operations. Power distribution, internal failure sensing, switching, and logic are also included in the power system.

In addition to powering all MCM systems, the MCM power system is also capable of supplying power to all other modules of the MSSR spacecraft system when required. In its normal operation, the power system trickle charges the MAV battery which not only powers the systems on that vehicle but also operates as part of the Mars lander module

(MLM) system power and is used to trickle charge the ERV battery which, in turn, trickle charges the small battery on the Earth entry capsule (EEC) or Earth orbit capsule (EOC).

Pyrotechnics, Thermal Control, and Mechanical Devices

The pyrotechnic system provides the devices and electrical switching circuitry necessary for the actuation of propulsion system squib valves and other spacecraft squib-actuated devices as well as the capability for power conditioning, switching, and telemetry output. The system includes pyrotechnic devices, one pyrotechnics control unit, and one propulsion actuation unit. Pyrotechnic devices for the MCM include mechanisms to effect solar panel deployment, bioshield release, descent package release, and propellant isolation.

The function of the propulsion actuation unit is to operate the solenoid valves of the attitude propulsion system. In addition to providing a power-switching function, it incorporates logic to hold the motor control valves open for the time interval between discrete "open" and "close" commands from the MCM attitude propulsion system electronics (for control of the twelve small thrusters) and from the MCM data handling and command system (for control of the rocket motor).

The pyrotechnics control unit operates all squib-actuated devices on the MCM on command from the data handling and command system.

The MCM temperature control system comprises thermal blankets, louvers, and heaters required to maintain all sections of the MCM at their proper temperatures.

The MCM mechanical devices system comprises the release mechanisms for the bioshield and Mars entry capsule, solar panel and high-gain antenna deployment mechanisms and dampers, and the separation-initiated timer and pyro-arming switch as commonly used on Mariner spacecraft.

DESIGN AND OPERATION OF MARS ORBIT VEHICLE SYSTEMS

During Earth-to-Mars cruise, the MOV performs essentially the same functions as the MCM does for the baseline mission. In addition, the MOV must insert itself and the ERV into orbit (direct entry) or insert itself, the ERV, and the MEC into orbit (for the out-of-orbit entry mode). It must also perform station-keeping functions during phase-one orbiting and must provide propulsion and systems necessary for assisting in rendezvous and docking with the MAV and transfer of the sample to the ERV. The MOV must also aline and spin up the ERV for Mars orbit departure. A summary of systems and masses for the 5 MOV variations is given in table VIII and a sketch of the MOV is shown in figure 13.

TABLE VIII.- SUMMARY OF SYSTEMS AND MASSES FOR MARS ORBIT VEHICLES

System	Mass, kg				
	$(MOV)_1$	$(MOV)_2$	$(MOV)_3$	$(MOV)_4$	$(MOV)_5$
Structure	59.9	59.9	59.9	59.9	59.9
Telecommunications	17.7	17.7	17.7	17.7	17.7
Power	23.6	23.6	23.6	23.6	23.6
Data handling and command	2.3	2.3	2.3	2.3	2.3
Pyrotechnics	7.3	7.3	7.3	7.3	7.3
Cabling	18.6	18.6	18.6	18.6	18.6
Temperature control	16.3	16.3	16.3	16.3	16.3
Mechanical devices	15.4	15.4	15.4	15.4	15.4
Articulation control	5.9	5.9	5.9	5.9	5.9
Rendezvous and docking	20.0	20.0	20.0	20.0	20.0
Propulsion and attitude control	621.9	640.7	616.7	2534.9	1873.4
Total	808.9	827.7	803.7	2721.9	2060.4

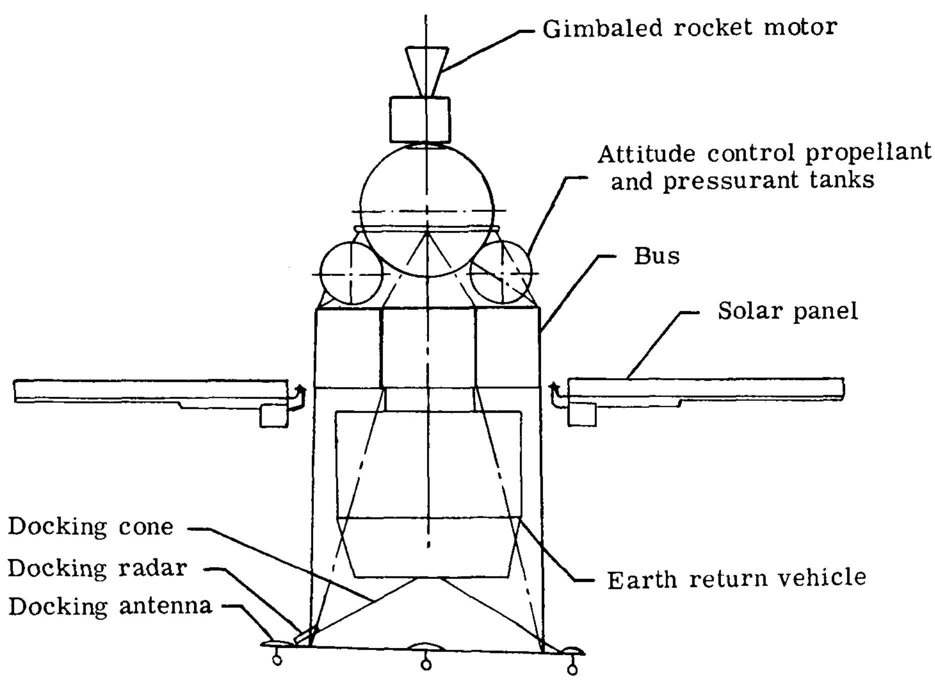


Figure 13.- Sketch of Mars orbit vehicle.

Propulsion and Attitude Control

The propulsion and attitude control system is essentially identical in design to that employed on the MCM. The variations in propulsion and attitude control systems masses between the five MOVs are due to differences in the masses of the MECs and ERVs which the MOVs support and the two Mars entry modes. (See table VI.)

Telecommunications, Data Handling and Command

The MOV telecommunications system is identical to that on the MCM. The system mass is slightly greater than that on the MCM because the larger spacecraft size demands somewhat longer antenna cables. In addition to the functions which the MOV provides during Earth-to-Mars cruise and during phase-one orbiting on the Mars direct entry missions, the MOV on the out-of-orbit entry missions receives the command which initiates the separation and deorbit maneuvers of the MEC.

The MOV data handling and command (DHAC) system is only a partial subsystem since it operates as a satellite of the DHAC system on the ERV which controls it, although its power is supplied by the MOV. It consists of two redundant command output units which act on a coded command word sent from the ERV DHAC system. Commands, then, are received by the MOV radio and sent to the ERV DHAC system. When a command is intended for the MOV, the DHAC system of the ERV sends the properly addressed coded command word to the satellite subsystem on the MOV. The MOV DHAC system further consists of a nonredundant data block which comprises an interface unit, an analog multiplexer, an analog-to-digital converter, a data memory, and an input/output control and digital formatter. These elements are identical to those used in the ERV DHAC system. All MOV data are digitized in this data block and formatted so that they can be sent to the DHAC system data formatter on the ERV as a single digital data stream. Combined with ERV data, the MOV data are then sent to the MOV radio for transmittal to Earth. For redundancy purposes, a few important measurements are also sent directly to the ERV DHAC system.

Associated with the DHAC system is a small solid-state spinup and release event timer which performs the control functions (including ERV spinup and release) required on the MOV after the cable interconnecting the MOV with the ERV has been separated. This cable separation starts the timer.

Structure

The MOV structure system consists of the bus and electronics support structure as well as supports for propellant and pressurant tanks, telecommunications antennas, solar panels, docking adapter, Mars entry capsule, bioshield, and ERV. Also included are release guides for the ERV. The ERV support is mounted to the ERV spin table. The docking adapter support also supports the rendezvous and docking system support ring.

Power

The MOV power system is identical to that on the MCM, except that the solar panels are articulated and slightly larger and the dc/ac/dc converters have less output power as given by reduced system demands for such regulated power. The functions, including supplying power to other modules when needed, are the same as those described for the MCM power system.

Pyrotechnics, Temperature Control, and Mechanical Devices

The MOV pyrotechnics system is identical to that on the MCM except that additional pyro devices are used on the MOV to separate the docking cone, the ERV (from its support on the spin table), and the cable interconnecting the MOV with the ERV. The MOV temperature control subsystem comprises thermal blankets, louvers, and heaters. The MOV mechanical devices system consists of mechanisms for release of the MEC, bioshield, MOV and ERV interconnecting cable, ERV, docking cone, docking cone cable cutter, spin table and its drive, separation-initiated timer and pyro-arming switch which both operate shortly after launch, and the deployment mechanisms and dampers for the solar panels.

Rendezvous and Docking Systems

The MOVs for all missions have the same rendezvous and docking systems. They play different roles, however, between the missions employing a three-axis controlled second stage on the Mars ascent vehicle (MAV) and the missions employing a spin-stabilized second stage on the MAV. The system includes the docking cone assembly, the approach guidance radar with four antennas and their feeds, two X,Y optical trackers, gallium-arsenide (GaAs) light source assembly, and the rendezvous and docking interface electronics. Rendezvous and docking systems required on the MAVs are discussed in the section "Analysis, Design, and Operation of Mars Ascent Systems."

RENDEZVOUS AND DOCKING OPERATIONS

Missions Employing MAV With Three-Axis-Controlled Second Stage

Prior to start of the rendezvous phase, the MAV is in an orbital cruise mode: a two-way communications link with Earth is maintained, the MAV attitude being locked to the Sun and any roll torques being controlled inertially; power is obtained from the four solar panels. Rendezvous operations then begin and continue with a number of maneuvers during each of which the MAV attitude is fully inertially controlled, but with the cruise mode reestablished after each maneuver. During all cruise-mode periods, the MAV is being tracked from Earth, as is the MOV (with which it is to dock), which maintains independent two-way communications with Earth.

The rendezvous philosophy demands that the MAV perform only orientation; it is the MOV which, by ground commands, maneuvers to effect the rendezvous, that is, the MOV flies to the MAV. Tracking and orbit determination continue throughout the rendezvous phase for both vehicles.

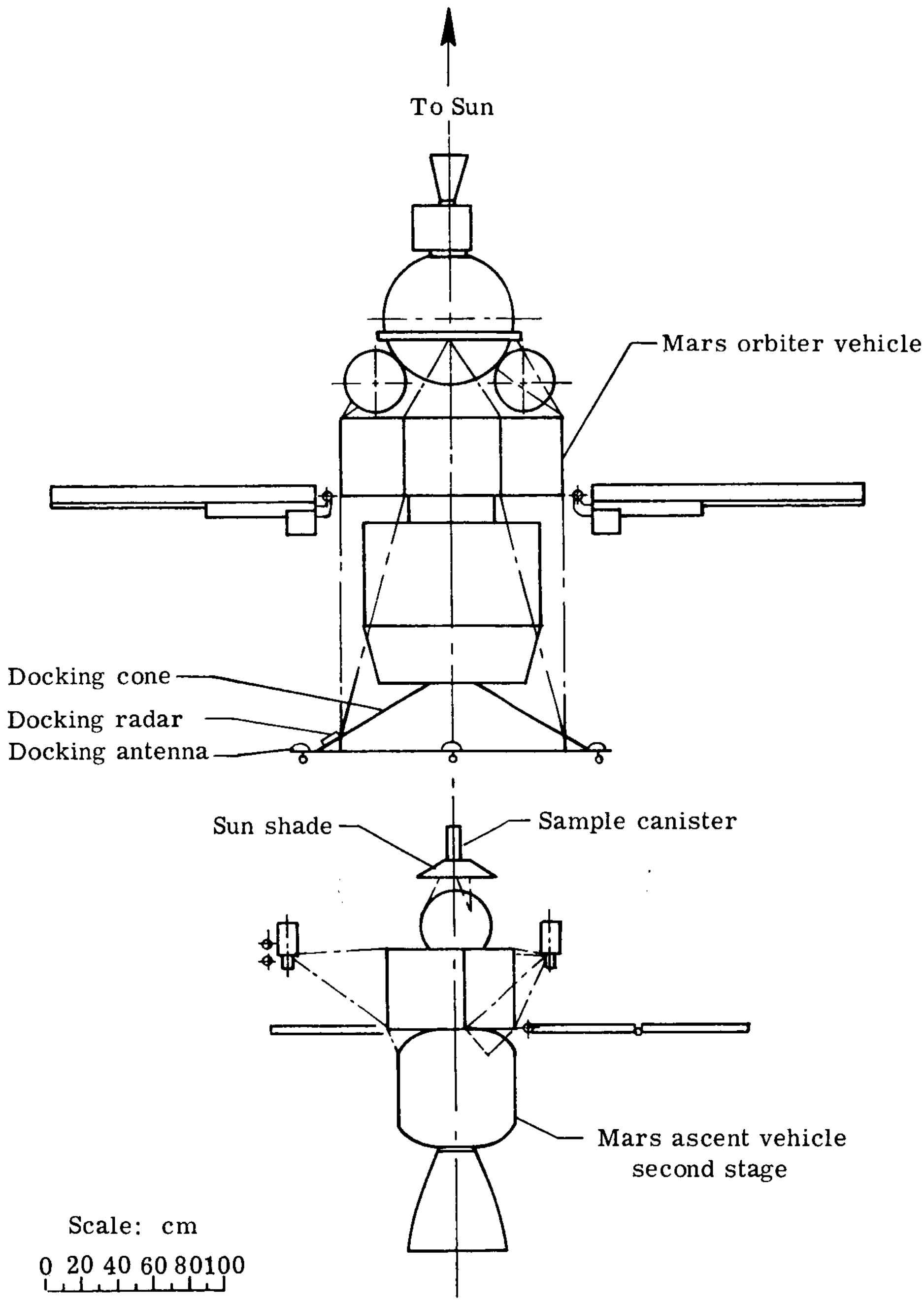
The approach guidance radar on the MOV is turned on when the two vehicles have reached about 30 km. This radar then takes over control of the MOV, and maneuvers are performed by the MOV in accordance with radar outputs. The rendezvous and docking operations are intended to be entirely automatic; however, ground personnel have override capability, by means of commands, and monitor data continuously from both vehicles. Rendezvous is continued in this manner until the station-keeping point is reached, at approximately 10 m, with the MOV positioned above the MAV, and with both vehicles attitude-referenced to the Sun. A status check is performed at this point.

If the status is verified to be normal, the final approach and docking operations begin. The modulated light source on the MAV is turned on, as are the two X,Y optical trackers on the MOV; MOV guidance is now controlled by the optical trackers. MAV guidance is switched to three-axis inertial (unless an automatic switchover has occurred at an earlier point because of MAV Sun-sensor shading by the MOV). The relative position of the two vehicles at this time is as shown in figure 14(a).

If the status check reveals an anomaly in the orientation of the MAV, the MAV can be commanded from the ground to use the sensors, responding to the MOV GaAs light source for closed-loop pointing of the MAV, in conjunction with the X,Y trackers on the MOV, so that the vehicles are able to point at each other for final approach and docking.

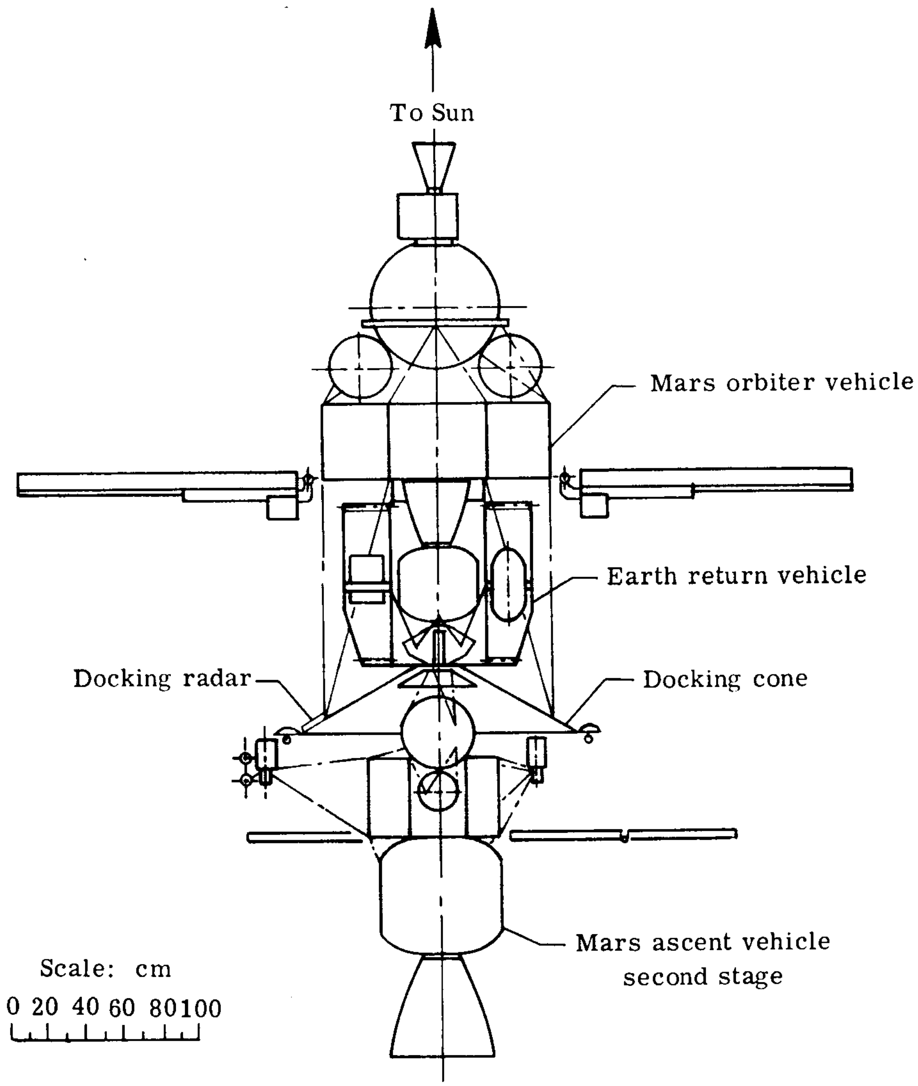
The MOV then performs its final maneuvers toward the MAV until the two vehicles are docked and latched. (See fig. 14(b).) After docking is verified, the MAV canister insertion ram is actuated and the canister is moved into the canister compartment on the Earth orbiting (or Earth entry) capsule. A small cavity in the canister is designed to take a spring-loaded temperature transducer (probe) which is integral with the canister compartment and thereafter provides canister thermal data from transmission by means of the ERV telemetry link. The canister compartment is then sealed. Canister installation is verified by the MOV telemetry. All data (including those handled by the ERV prior to ERV separation) are transmitted by the MOV radio.

The MAV is now ready for undocking from the MOV. Upon command, the docking cone of the MOV (which is still latched to the MAV docking adapter) is separated from the MOV, together with its structure and the rendezvous docking system on the MOV. (See fig. 14(c).) Separation springs impart a small velocity increment to the MAV.



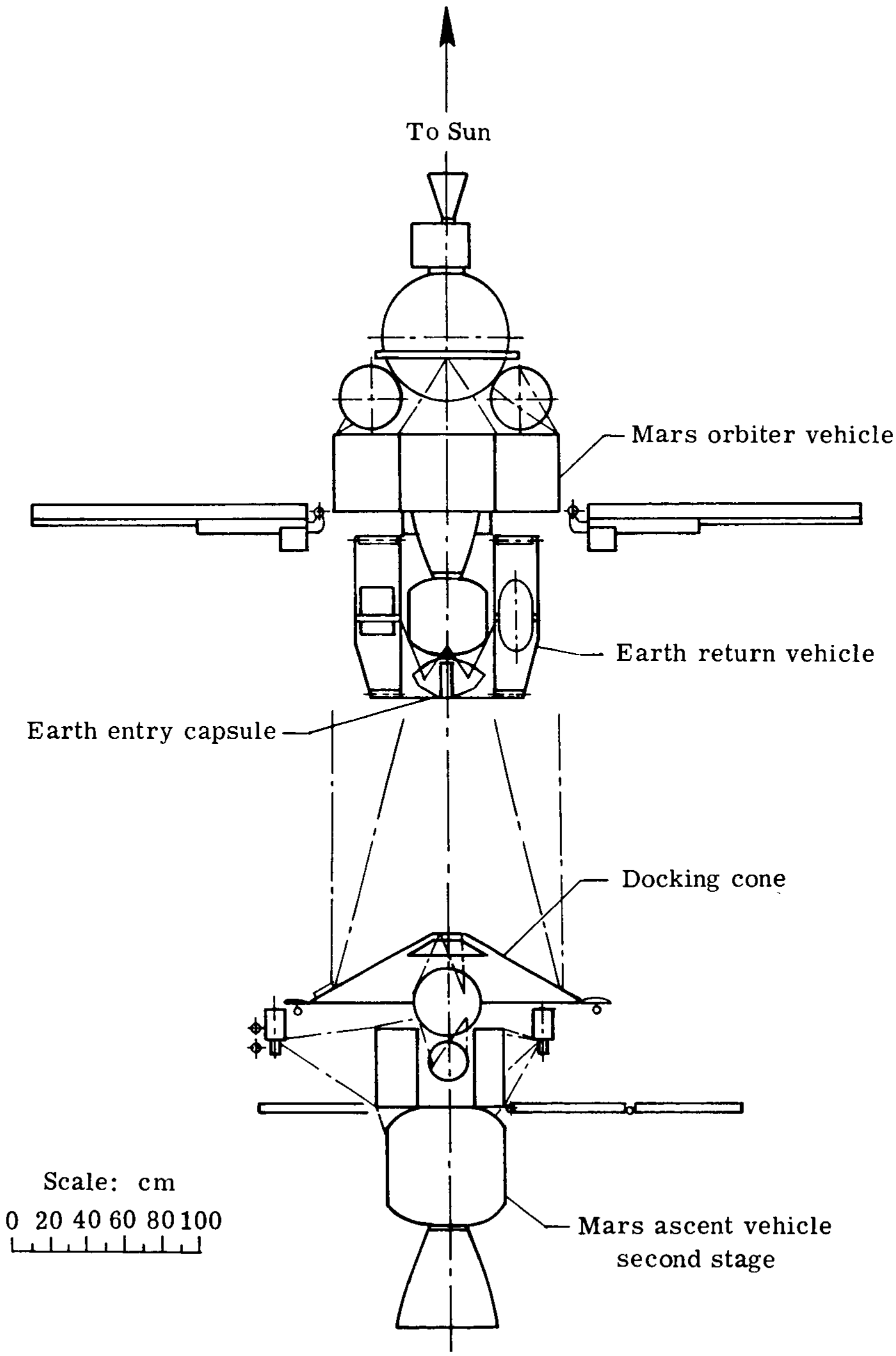
(a) Position prior to docking.

Figure 14.- Illustration of docking positions.



(b) Docked position.

Figure 14.- Continued.



(c) After undocking.

Figure 14.- Concluded.

Spring-imparted separation places the MAV a sufficient distance from the MOV to permit the MOV to reorient itself for separation from the ERV.

Missions Employing MAV With Spin-Stabilized Second Stage

For these missions the MAV does not dock with the MOV. When the MOV has maneuvered to within about 3 meters of the MAV, the MAV will be despun, and the canister will be separated and launched very slowly toward the MOV. The MOV will capture the canister in the docking cone and, by maneuvering, force the canister along the cone inner surface to and into the canister compartment in the EEC or EOC. The sample canister will be sealed inside the canister of the EEC or EOC in the same manner as on the other MOR missions.

The rendezvous operations are nearly identical to those for the baseline mission. The MAV attitude is fixed inertially, however, and the MOV will not attempt to aline its axes with those of the MAV until the MOV is about 5 meters from the MAV and almost directly in front of it. At a separation distance of about 3 meters, the MOV will stop and verify with Earth that it is properly alined with the MAV. When attitude is verified, the MAV will be despun and the canister will be separated and launched very slowly toward the MOV.

ANALYSIS, DESIGN, AND OPERATION OF MARS ENTRY AND LANDING SYSTEM

Mars entry and landing system concepts were derived to decelerate the modules and vehicles contained in the Mars entry capsule (see fig. 9) safely to the surface. For the baseline MEC configuration, shown in figure 9(a), an ERV and EEC (or EOC) must be carried to the surface as well as an MLM and MAV. In contrast, the MOR/MEC configuration, seen in figure 9(b), contains only the MLM and a smaller version of the MAV; the ERV and EEC (or EOC) are located in the Mars orbiter vehicle where they remain until MOV/ERV separation. Thus, the baseline MEC configuration is the more massive of the two types and the more demanding on entry and landing deceleration systems.

Five entry and landing system concepts were derived to meet a range of entry conditions and landed mass requirements. One concept was derived for the baseline missions and served as a base for the modeling of the other concepts. The baseline concept is used for direct Mars entry at 5.5 km/sec and delivers the maximum mass to the surface of all concepts. Four concepts of decreased capability were derived to meet the particular requirements for the MOR missions. These requirements included both direct and out-of-orbit Mars entries.

The Viking '75 entry technology base was drawn on heavily for the MSSR entry and landing system designs; however, significant changes from that technology, both in system designs and operations, were made, where necessary, to decelerate the much greater MSSR entry masses. This approach was preferred over the development of new systems requiring altogether new technology.

For all five MSSR concepts Viking-type aeroshells, parachutes, and terminal descent motors were used. However, the aeroshell was strengthened and heat protection was added. The parachute diameter was increased and a pilot parachute was used for main parachute deployment for the three most massive MSSR concepts. These also employed an increased Mach number and dynamic pressure for the main parachute deployment point. Additional terminal descent rocket motors were used for all five entry and landing concepts.

In the baseline MSSR entry, the aeroshell decelerates the MEC to a Mach number of about 3 where the pilot parachute is mortar-deployed. The pilot parachute then deploys the main parachute at a Mach number of about 2.3 and at $q = 575\ N/m^2$. The parachute system slows the MLM/MAV until an altitude of about 1 km where nine terminal descent motors ignite to bring the MLM/MAV to a soft landing. No Earth communication takes place during entry or landing. The Viking mean atmosphere model (fig. 5) was used along with a wind model for designing all concepts.

The Viking entry body diameter of 3.51 m has been used in this report for entry analyses. However, the Titan/Centaur shroud can accommodate MSSR spacecraft up to 3.71 m in diameter and the Shuttle/Centaur shroud can accommodate MSSR spacecraft as large as 4.27-m diameter. (See fig. 12.) The effect of these increased diameters has been considered briefly as a matter of interest for missions in the 1980's. The Titan/Centaur and Shuttle/Centaur Earth-to-Mars payload injection mass capabilities were shown in figure 11.

This section of the report will discuss the analyses which led to formulation of the baseline concept and the four variations for the MOR missions. The concepts are presented as well as their companion entry trajectories.

ENTRY TRAJECTORY AND DECELERATOR ANALYSIS

The five Mars entry capsules must land integrated Mars launch configurations of three different masses and, except for the baseline concept, incorporate the use of both direct and out-of-orbit entry modes (5.5 km/sec and 4.6 km/sec, respectively). Therefore, each MEC was assumed to require a different entry mass and decelerator system capability. Entry mass and decelerator designs were analyzed for the baseline mission first, since this mission required both maximum entry mass and maximum deceleration capability.

The analyses were conducted in three phases corresponding to the aeroshell, parachute, and terminal descent phases of deceleration. The Viking '75 Mars mean atmosphere model was selected for the analyses from the three current models. (See fig. 5.) This choice was made after comparing entry vehicle velocities, near the altitude of terminal descent initiation, in each of the three model atmospheres. Each of the three runs used identical vehicles and V_E and γ_E values, and covered only the aeroshell and parachute phases of deceleration, which depend on the atmosphere. The vehicle's velocity dispersion between the maximum ρ_s and minimum ρ_s models was found to be small (30 m/sec) for purposes of this study, and because the mean atmosphere model produced a mean velocity result at the terminal descent initiation (TDI) point, the mean atmosphere was selected for use in the entry analyses. It will be seen later that the effect of the three atmosphere models on landed mass was also assessed and the additional entry mass needed for the "worst case" atmosphere model was derived. A wind model shown in figure 15 was used for all entry analyses.

Aeroshell Phase

The aeroshell phase of entry begins at initial atmospheric entry (an altitude of 244 km) and extends to an altitude of about 6 to 12 km, where Mach number and dynamic pressure are reduced below preselected values so that the parachute can be deployed to initiate the next phase of entry.

Parameters affecting the entry body's mass-carrying capability and its entry profile during the aeroshell phase are V_E, γ_E, L/D, C_DA, and the parachute deployment conditions, M_d and q_d. In these analyses, a fixed V_E, C_DA, and maximum M_d and q_d values were used so that the maximum M_E was determined as a function of aeroshell L/D and entry angle. As discussed in the section "Trajectory Analysis and Design Parameters," the direct entry velocity at Mars for 1979 is about 5.5 km/sec and the targetable entry angle uncertainty is $\pm 1^{\circ}$. For out-of-orbit entry, the Viking '75 entry velocity of 4.6 km/sec was used, and based on current assessments for a tentative 1979 Viking mission (ref. 3), a conservative value of $\pm 1^{\circ}$ was also chosen for the out-of-orbit entry angle uncertainty.

<u>Determination of maximum values of M_d and q_d during aeroshell phase</u>.- The maximum M_d and q_d are the terminal conditions to be met by the aeroshell decelerator before the parachute can be safely deployed. They are therefore significant factors in determining aeroshell phase $M_{E,max}$. Higher M_d and q_d values allow higher $M_{E,max}$ but produce more stringent operating conditions on the parachute at its deployment. To determine the highest feasible M_d and q_d values for MSSR entries, early

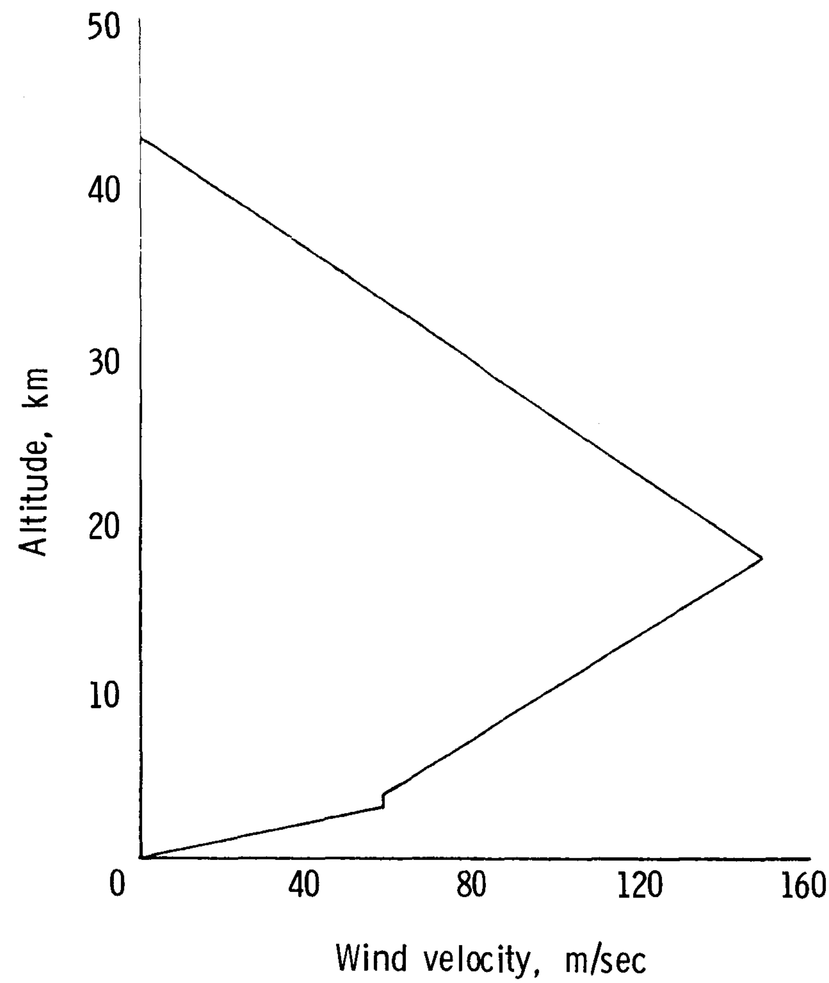


Figure 15.- Tail-wind model profile for Mars entry analysis.

Mars parachute flight test data (refs. 5 to 9) and Viking '75 final test data (ref. 10) were considered. Early flight test data showed successful deployments at Mach numbers up to 33 and q_d up to 555 N/m^2, and up to 929 N/m^2 with some canopy damage. The Viking parachute, originally designed for Mars mortar-fire deployment conditions of $M_d \leqq 2.2$ and $q_d \leqq 431$ N/m^2, sustained canopy damage at Earth flight test values of $M_d = 2.1$ and $q_d = 565$ N/m^2. Failure at this point was thought to be mainly due to asymmetric canopy loading at first inflation caused by a small angle of attack at mortar firing. The MSSR main parachute can be preceded by a pilot parachute which will tend to aline the main parachute with the flow field and reduce the opening load. The main parachute could also be expected to use improved canopy fibers now available. As a result, for the three most massive MSSR entry concepts, the values $M_d \leqq 2.3$ and $q_d \leqq 575$ N/m^2 were chosen. For the other two concepts, earlier Viking '75 deployment values of $M_d \leqq 2.2$ and $q_d \leqq 431$ N/m^2 were selected.

The technique used to define the maximum M_E required several steps. A set of aeroshell phase entries were made by covering a range of likely entry angles but by using one value of M_E and one L/D value. The drag of the pilot parachute being neglected, those runs in which the aeroshell was able to decelerate to $M_d \leqq 2.3$ and $q_d \leqq 575$ N/m^2 by an altitude of about 6 km were admissible and the others were rejected as having entry angles which were too steep or too shallow. The 6-km altitude level was assumed as the minimum required for parachute and terminal descent deceleration. If greater than a 2° band of entry angles were admissible, the M_E was increased by 44-kg unit increments and another set of runs made. If the band of admissible entry angles became less than 2°, the L/D value was raised in 0.02 steps until a maximum M_E value was determined.

Results from the many entry simulations made to obtain the maximum M_E are shown in figure 16. The solid points illustrate the final values as derived to the nearest 44 kg of mass and 0.02 units of L/D for the two entry velocities (5.5 km/sec and 4.6 km/sec) and $\Delta\gamma_E \approx 2^{\circ}$. The values at these points are 2858 kg and 2903 kg. The

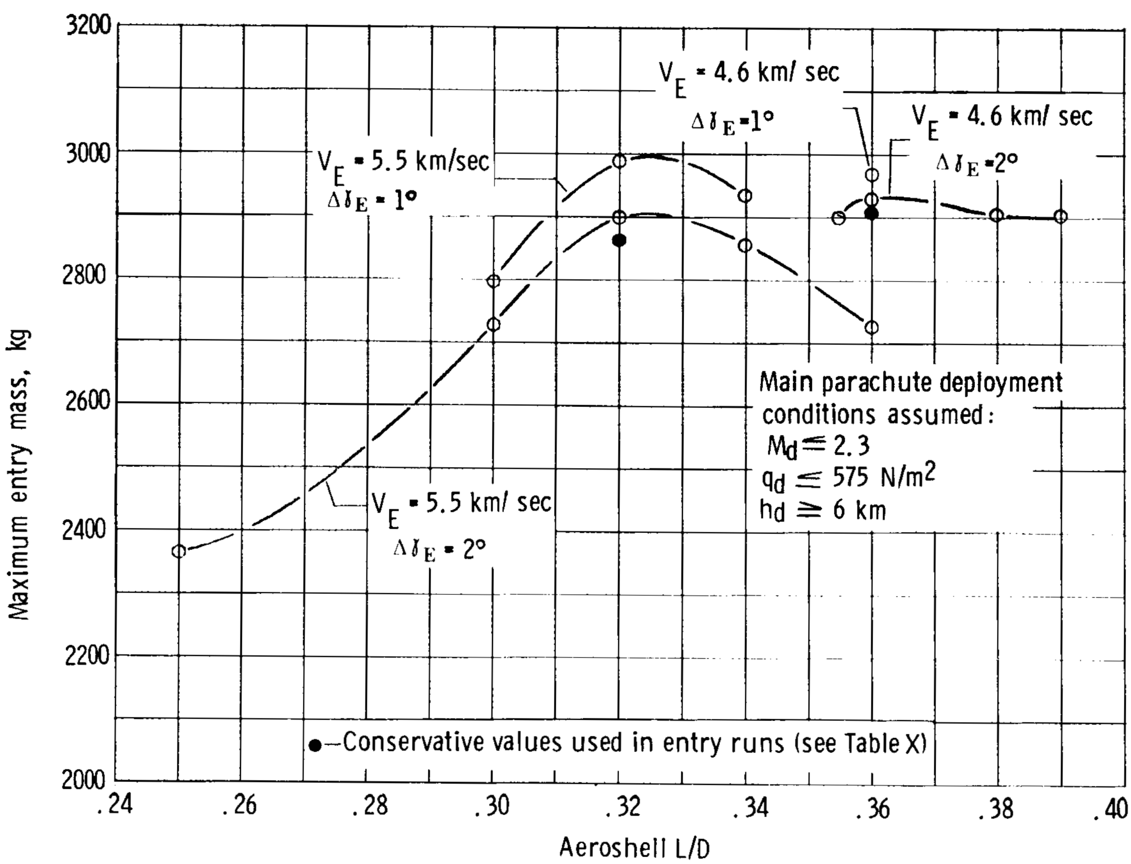


Figure 16.- Effect of aeroshell L/D on maximum Mars entry mass at two entry velocities and two entry corridor widths using the mean atmosphere model.

faired curves were derived by graphical interpolations of the simulated entry data showing the exact $M_{E,max}$ values at $\Delta\gamma_E = 2^o$ (and at $\Delta\gamma_E = 1^o$ for comparison). For purposes of entry concept designing, the solid point values taken directly from the data were used, not the idealized interpolated values. Table IX summarizes the entry parameters which resulted from the use of the 2858-kg and 2903-kg values.

TABLE IX.- MARS ENTRY PARAMETERS FOR MAXIMIZED ENTRY MASS CASES

[Use of the Mars mean atmosphere model assumed]

Entry parameter	Entry mode	
	Direct	Out of orbit
Entry velocity, km/sec	5.5	4.6
Entry angle corridor, deg	-26.8 to -24.8	-26 to -24
Entry mass for 3.51-m-diameter aeroshell, kg	2858	2903
L/D (hypersonic).	0.32	0.36
C_D (hypersonic)	1.44	1.39
Trim angle of attack, deg.	-19.8	-22.2
$(M_E/C_DA)_{max}$ (hypersonic), kg/m^2	205	216

Entry mass capability for various diameters.- By using $(M_E/C_DA)_{max}$ values given in table IX, maximum M_E as a function of the diameter was determined. The Viking '75 aeroshell diameter (3.51 m) has been taken as the baseline in this study, but consideration was given to the mass resulting from the use of larger diameters. Two increased diameters of particular interest are the maximum which can be accommodated in the Titan/Centaur shroud (3.71 m), and the anticipated maximum diameter suitable for the Shuttle/Centaur (4.27 m) for use on missions in the 1980's. Figure 17 shows the variation of maximum entry mass with aeroshell diameter for the two $(M_E/C_DA)_{max}$ values, 216 kg/m^2 and 205 kg/m^2, previously derived. For comparison, a third curve based on the $(M_E/C_DA)_{max}$ of 199 kg/m^2 derived for $V_E = 6.4$ km/sec is included. This velocity is slightly greater than the maximum values for direct entry missions in the 1980's.

Aerodynamic loading and heating during aeroshell phase.- Steeper (more negative) entry angles and higher values of L/D than those required for Viking '75 resulted in higher aerodynamic pressure, loading, and stagnation-point heating during MSSR entries. Figure 18 shows maximum values of these entry quantities as a function of M_E/C_DA which might occur in the ranges of γ_E and L/D indicated. Heating was assumed to be negligible below an altitude of 12 km, about the altitude of parachute deployment. Changes in these quantities with entry velocities are also shown in figure 18.

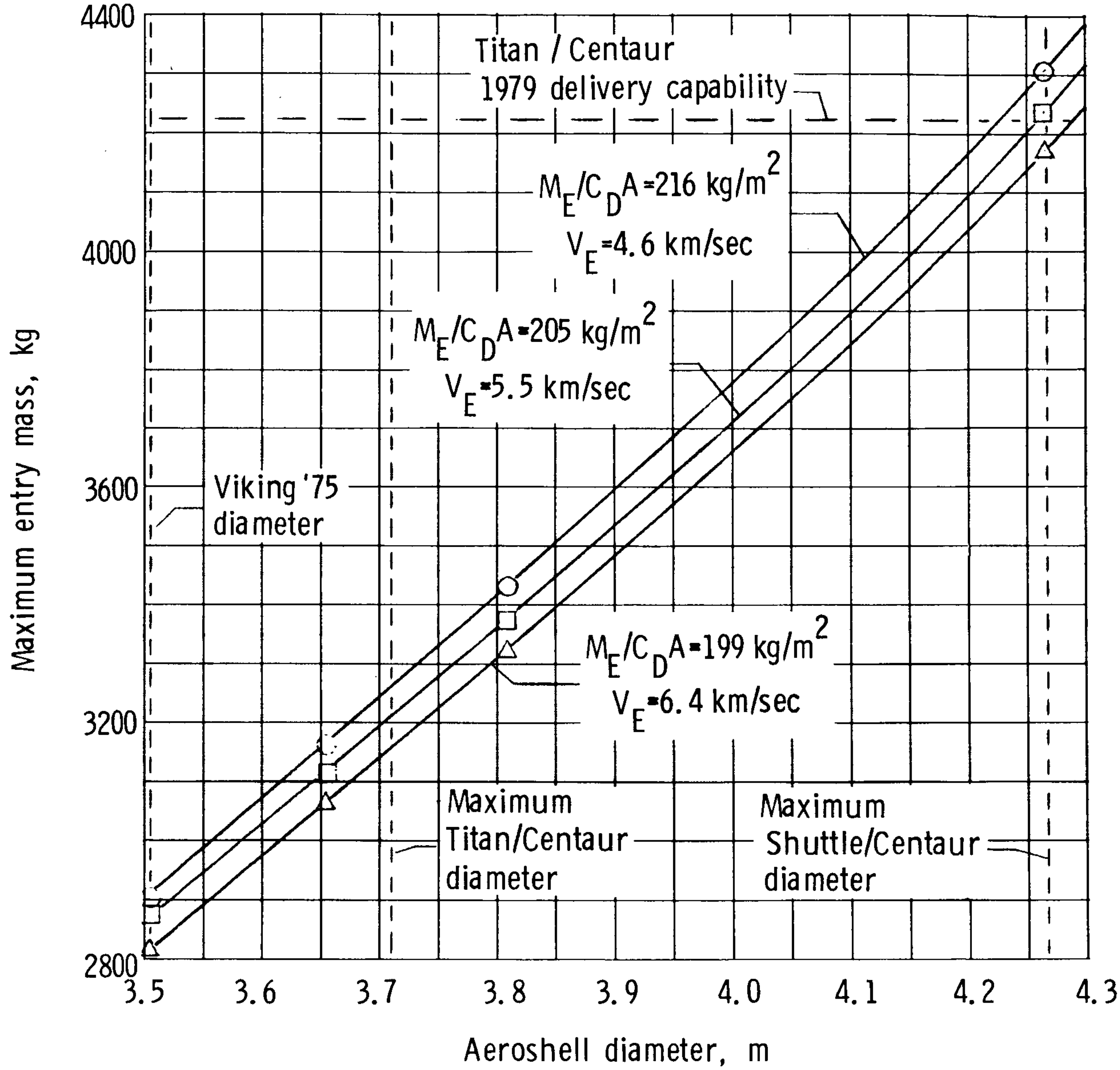


Figure 17.- Variation of maximum entry mass with aeroshell diameter for three maximized M_E/C_DA conditions and entry velocities in the Mars mean atmosphere model.

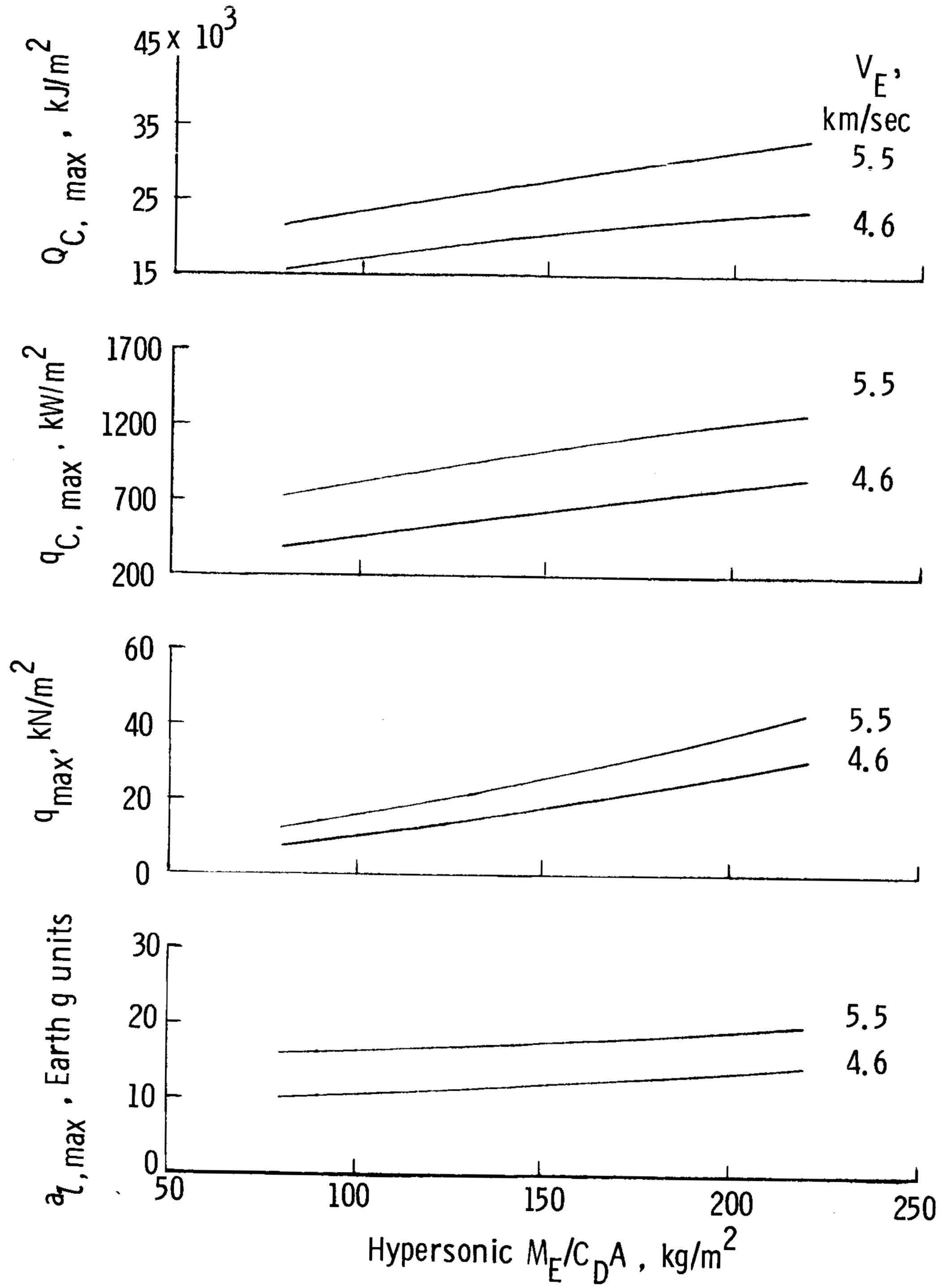


Figure 18.- Maximum Mars entry parameters for γ_E of -18° to -27° and L/D of 0.18 to 0.36.

Parachute Phase

Mortaring of larger parachutes posed severe structural and volumetric problems for the MEC designs (refs. 11 and 12), particularly at the higher angles of attack used for attaining high L/D values. Therefore, a pilot parachute deployment technique was selected to deploy the main parachute for the three most massive MEC concepts. This deployment concept was used in the study of the performance of the larger diameter parachutes. In this technique the parachute phase is initiated with a 6-meter-diameter mortar-ejected pilot parachute at a Mach number of approximately 3.0. This small parachute is then used to pull out the main parachute when the proper conditions of M_d and q_d are reached. This pilot parachute tends to aline the main parachute with the flow stream and minimizes the angle-of-attack problem at the beginning of inflation. The large main parachute can be packed in a location of convenience, other than on the center line. However, this technique requires careful designing and testing to insure that the proper pilot parachute suspension line lengths are used to keep it filled while it is in the wake of its main parachute.

Main parachute performance analysis.- The performance of three increased-diameter Viking '75 type parachutes was analyzed (Viking parachute diameter is 16.1 m). These three parachutes, 19.8 m, 22.9 m, and 25.9 m in diameter, used suspension-line lengths derived from the Viking '75 optimum payload-parachute geometry $\left(\text{Length} = 1.7D_o\right)$. No riser line was used and the payload bridle and swivel were eliminated by fastening the suspension lines in clusters directly to a small circle of load-carrying points symmetrically located on the MEC base cover.

To compare the performance of the three candidate parachutes, aeroshell and parachute phase entry simulations were made by using various entry masses and entry parameters. All simulations were terminated at an altitude of about 1.5 km where terminal descent would be expected to follow. One entry set, demonstrating the maximum entry mass case for $V_E = 5.5$ km/sec shown in table IX, is presented in figure 19. The effects of the simulated surface winds were included in the parachute runs.

Terminal Descent Phase

The terminal descent phase of entry and landing begins with the ignition of the retrothrust engines at a preselected altitude above Mars mean surface level and concludes with the touchdown of the MLM and MAV combination (and the ERV and EEC or EOC, for baseline missions). The Viking system provides the best descent system technology available for Mars landings but increased thrust capability had to be provided for terminal deceleration for the MSSR vehicles. The solution pursued was to use more Viking motors and to preserve the control technology. The use of six, nine, and twelve Viking '75 motors in three clusters was investigated. The motor masses and the system plumbing mass

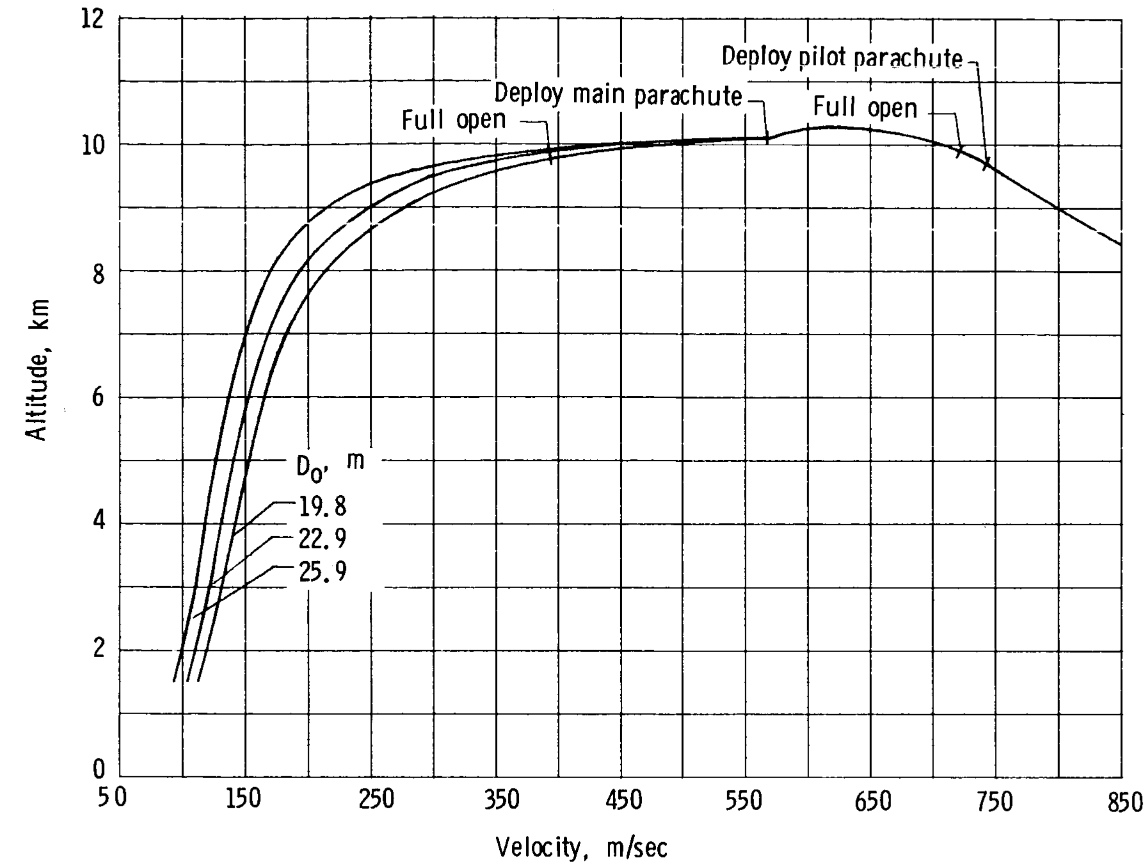


Figure 19.- Parachute-phase velocity profiles for three candidate parachute sizes using the 5.5-km/sec (direct) entry conditions of table IX.

requirements were modeled after those of Viking '75. Tankage mass was assumed to be 18 percent of the total propellant mass carried and a 5-percent residual propellant mass was assumed to remain at shutdown.

Terminal descent calculations were made by using a given initial velocity, vehicle mass, flight-path angle, and total propellant mass to determine the propellant consumption and altitude segment required to bring the MLM/MAV to touchdown. Sets of these calculations were generated by varying the initial velocity and vehicle mass at thrust initiation in step fashion. This was done for six, nine, and twelve motor combinations. For the maximum entry and landed mass case (baseline type), the twelve-motor combination proved to be the most efficient because of its highest ratio of thrust to mass. However, twelve motors proved to be too difficult to incorporate in the volume and structural limitations of the MLM design. Therefore the nine-motor combination was selected for the baseline mission, and six motors were found to be adequate for the MOR missions.

Parachute and Terminal Descent Phase Matching

Final selection of the best parachute diameter depended on proper matching with the terminal descent system. Complete entry runs were used with varying system combinations for each phase. For the baseline mission initially, and later for the MOR missions, the effect of entry velocity, parachute diameter, and number of terminal descent motors on usable landed mass was determined. Usable landed mass is defined as the total landed mass minus the mass of remaining descent propulsion inerts, residuals, and pressurant.

Figure 20 shows the effect of both parachute diameter and entry velocity on usable landed mass, for the maximized entry cases of table IX. Table X shows a breakdown of final results of usable landed mass for the three parachutes when used for the baseline mission. The 22.9-m-diameter parachute was selected for the baseline mission since it results in essentially equal usable landed mass with the largest parachute but requires less volume.

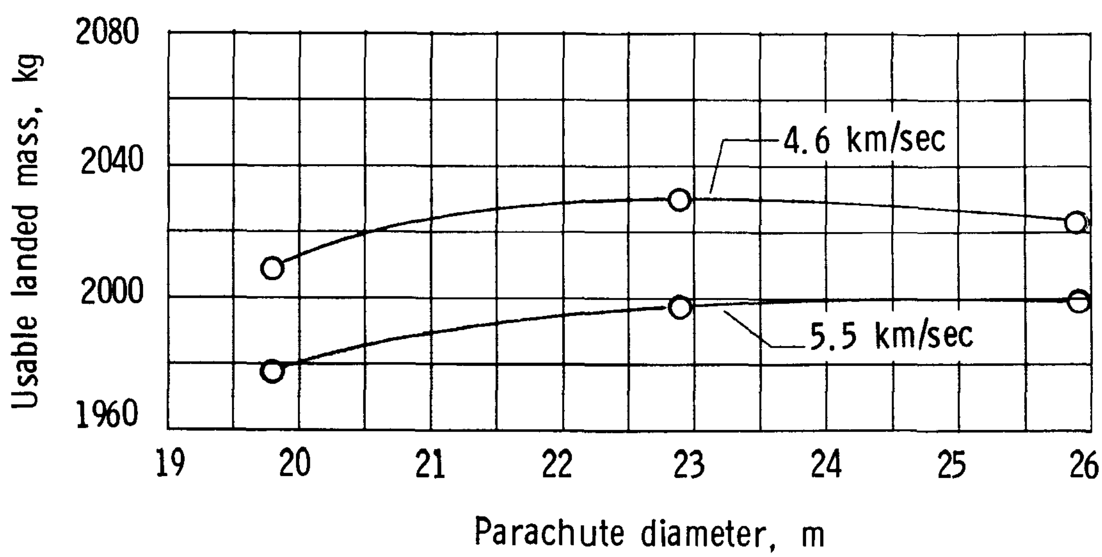


Figure 20.- Effect of parachute diameter and entry velocity on usable landed mass. The two maximized entry mass conditions shown in table IX and the use of nine engines on terminal descent are assumed.

MOR Mission Summary

Entry mass requirements for the MOR missions were based entirely on the need to land either of two MAVs whose masses were 557 kg and 1071 kg for a spin-stabilized design and a three-axis stabilized design, respectively. The MOR mission entry parameters derived are presented in table XI along with those presented earlier for the baseline mission.

TABLE X.- EFFECT OF PARACHUTE DIAMETER ON USABLE LANDED MASS CAPABILITY FOR BASELINE ENTRY MASS CASE

[Mars mean atmosphere model is assumed]

Entry parameter	Values for parachute size, D_o, m, of –		
	19.8	22.9	25.9
Entry velocity, km/sec	5.5	5.5	5.5
Entry mass, kg (3.51-m diam)	2858	2858	2858
Altitude at parachute deployment, km	10.1	10.1	10.1
Parachute phase mass, kg	2582	2582	2582
M_d	$\leqq$2.3	$\leqq$2.3	$\leqq$2.3
q_d, N/m^2	$\leqq$575	$\leqq$575	$\leqq$575
Mass at terminal descent initiation, kg	2385	2350	2303
Number of terminal descent engines	9	9	9
Velocity at terminal descent initiation, m/sec	$\approx$120	$\approx$105	$\approx$94
Altitude at terminal descent initiation, m	1920	1400	1040
Terminal descent propellant mass, kg	244	200	161
Mass at landing, kg	2141	2150	2142
Terminal descent propulsion inerts, residuals, kg	163	152	143
Usable landed mass, kg	1978	1998	1999

TABLE XI.- MARS ENTRY PARAMETERS FOR THE SELECTED MARS ENTRY CAPSULES

Entry parameter	Values for the Mars entry capsules				
	$(MEC)_1$	$(MEC)_2$	$(MEC)_3$	$(MEC)_4$	$(MEC)_5$
Entry mass, kg	2858	2086	1361	2086	1361
L/D (hypersonic)	0.32	0.26	0.18	0.28	0.18
Entry angle corridor, deg	-26.8 to -24.8	-24 to -22	-22 to -20	-22 to -20	-20 to -18
M_E/C_DA (hypersonic), kg/m^2	205	144	90	145	90
Entry velocity, km/sec	5.5	5.5	5.5	4.6	4.6

Parachute selection for MOR missions was based on the desire to use the Viking '75 parachute and deployment mode where possible and to use the baseline mission parachute and deployment in other cases. Therefore, the 16.1-m Viking parachute and deployment mode was used to decelerate the 557-kg MAV, which required a payload on the parachute about 55 percent greater than that for Viking '75, but the baseline mission parachute (and deployment mode with pilot parachute) was chosen for the 1071-kg MAV. The V_E and γ_E used for the individual MOR cases did not affect these choices since its effect on the parachute payload mass was very small.

Selection of the number of Viking '75 terminal descent motors required was based on landed mass delivery. For all the MOR missions, six motors were found to be adequate.

Atmosphere Model Effects On Landed Mass

At the beginning of the entry and landing analyses, the mean surface density atmosphere model was chosen for all the entry runs, but with the condition that the landed mass margin required to fly the worst case atmosphere model be evaluated later. In the later evaluations, using system masses which were preliminary for the baseline mission, complete entry runs were made in each of the three atmosphere models. The flight parameters used and the masses derived for each atmosphere model are tabulated in table XII. In the table, two sets of terminal descent propulsion requirements have been posed. Case 1 assumes that terminal descent propellant loading is based on the need to meet the "worst case" atmosphere condition with a 5-percent residual at touchdown. Further, the tankage mass was assumed to be 18 percent of the mass required for the propellant load.

The "worst case" atmosphere was the minimum surface density model which required 241 kg of propellant to be loaded and 230 kg consumed. This is an additional 30 kg of terminal descent propellant over that required for the mean atmosphere model and 66 kg above that required for the maximum ρ_s atmosphere. As a result the "worst case" landed mass is decreased from that for the mean atmosphere by 1.4 percent and the usable landed mass is reduced to 1975 kg for all atmospheres if the "worst case" condition is to be anticipated. For comparison, the usable landed masses which would have occurred if only the consumed terminal descent propellants (plus 5-percent residual) were aboard, are shown as case 2 in table XII. It can be seen that the "worst case" atmosphere produces a usable landed mass penalty of 1.9 percent compared with the mean atmosphere. As a result a minimum usable landed mass margin of 1.9 percent has been assumed to be necessary for all five MSSR entry concepts.

TABLE XII.- EFFECT OF MARS ATMOSPHERE MODELS ON TOTAL LANDED MASS AND USABLE LANDED MASS FOR PRELIMINARY BASELINE ENTRY

[Case 1 assumes loaded propellant and tankage necessary to fly baseline concept to surface in the worst case (minimum ρ_s) atmosphere

Case 2 assumes loaded propellant and tankage necessary to fly baseline concept to surface in each respective atmosphere]

Entry and landing parameter	Mars atmosphere model		
	Maximum ρ_s	Mean	Minimum ρ_s
Entry mass, kg	2858	2858	2858
L/D	0.32	0.32	0.32
Entry velocity, km/sec	5.5	5.5	5.5
Entry angle, deg	-25.8	-25.8	-25.8
Aeroshell diameter, m	3.51	3.51	3.51
Parachute diameter, m	22.9	22.9	22.9
Mass at terminal descent ignition (TDI), kg	2364	2364	2364
Case 1 –			
Terminal descent propellant loaded for all atmospheres, kg	241	241	241
Propellant consumed, kg	164	200	230
Total landed mass, kg	2200	2164	2134
Total landed mass variance (from mean atmosphere value, 2164 kg), percent	1.7	0	-1.4
Propulsion inerts and residuals, kg	225	189	159
Usable landed mass, kg	1975	1975	1975
Case 2 –			
Terminal descent propellant loaded, kg	172	210	241
Propellant consumed, kg	164	200	230
Total landed mass, kg	2200	2164	2134
Propulsion inerts and residuals, kg	143	152	159
Usable landed mass, kg	2057	2012	1975
Usable landed mass variance (from mean atmosphere value, 2012 kg), percent	2.2	0	-1.9

All the Mars entry capsules consist of three major elements: (1) a Viking-type aeroshell, (2) a parachute system, and (3) a lander module. The aeroshell provides initial aerodynamic deceleration to an altitude of about 10 km. Its attitude control system provides stability during this period and also provides the deorbit capability before entry for the missions employing out-of-orbit entry. At the required values of M_d and q_d, a parachute is deployed that decelerates the lander down to an altitude of about 1.5 km. During this period and during the terminal entry phase, the propulsion system on the MLM will provide attitude control. The same propulsion system will provide retropropulsion for a soft landing of the MLM. The MLM will contain the Mars ascent vehicle and will provide a launch platform for that vehicle. The MLM will also carry the sample-acquisition mechanism and all science-related instrumentation and equipment required during surface operations.

Deceleration requirements for the aeroshell, the parachute system, and the terminal propulsion system on the MLM, were determined from the trajectory and deceleration system analyses described in the previous section. Systems were derived for five Mars entry capsules. The systems for each capsule were selected to support a specific mission; that is, to land a payload of specified mass. The mass for the baseline mission was the greatest of the five because the landed mass includes the Earth return vehicle and EEC (or EOC). The four Mars entry capsules for the MOR missions include one for direct entry and one for out-of-orbit entry for each of the two types of Mars ascent vehicles. Table XIII summarizes the masses of the five MEC designs. For all five MEC concepts, some amount of usable landed "contingency" mass is available. For the out-of-orbit MOR three-axis case $(MEC)_4$, however, this contingency is not great enough to meet the 1.9-percent worst case atmosphere margin prescribed earlier. A small increase in the total mass of $(MEC)_4$ would correct this condition and can easily be accommodated by the entry systems involved. It will be seen, however, that the overall mass demands of the entry vehicle plus orbiter for this mission exceed the Titan/Centaur injection capability. The usable landed to entry mass ratio for the designs ranged from 0.64 to 0.70, the higher entry mass MECs being the more efficient. Views of two of the MEC designs are shown in figures 21 and 22.

All MEC designs (including MAVs) are sterilized to prevent contamination of Mars atmosphere or surface with terrestrial organisms. These designs considered the reduced capabilities of propellants and gases which were sterilized.

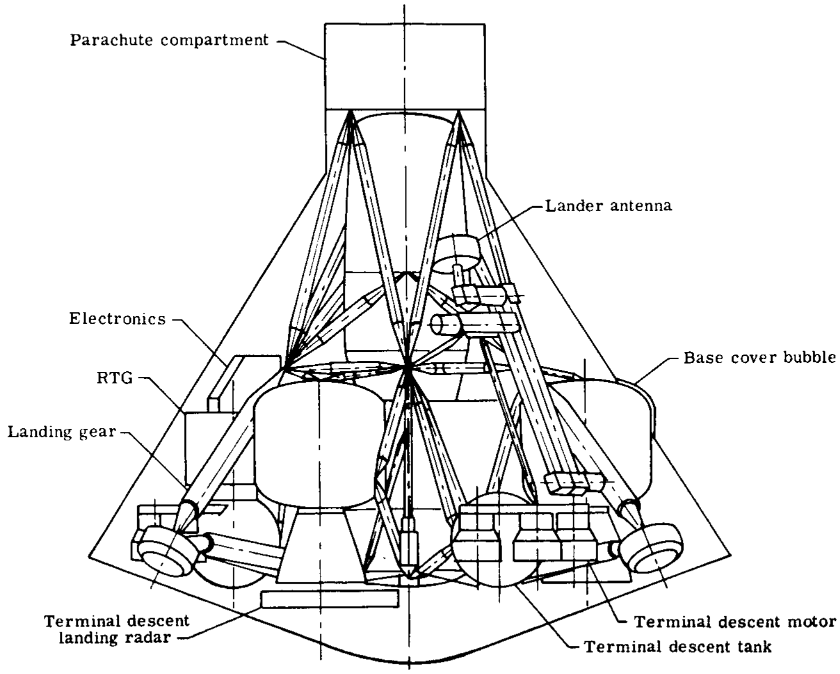
TABLE XIII.- MEC MASS SUMMARY

[See table VI for mission application]

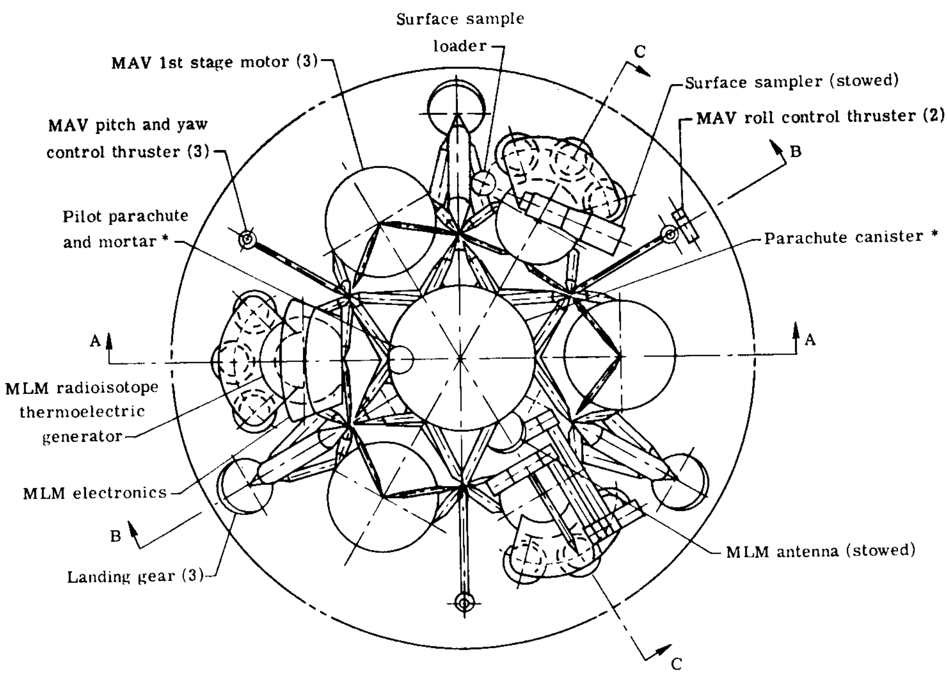
Item or event	Mass, kg, for each entry-departure mode, for –				
	$(MEC)_1$	$(MEC)_2$	$(MEC)_3$	$(MEC)_4$	$(MEC)_5$
Loaded, at separation from MCM/MOV	2858	2086	1361	2160	1410
Deorbit propellant used (ΔV = 75 m/sec)	-----	----	----	74	49
At entry	2858	2086	1361	2086	1361
Aeroshell (see table XIV)	276	237	200	[a]234	[a]185
After aeroshell separation	2582	1849	1161	1852	1176
Parachute systems (see table XV)	232	232	145	232	145
After parachute system separation	2350	1617	1016	1620	1031
Terminal descent propellant used	200	115	65	143	68
Total landed	2150	1502	951	1477	963
Terminal descent propulsion inerts, residuals	152	98	85	110	87
Usable landed (payload)	1998	1404	866	1367	876
MLM (without propulsion)	301	271	242	271	242
Balance available for launch configuration	1697	1133	624	1096	634
Required for launch configuration (see table XIX)	[b]1563	1071	557	1071	557
Contingency	134	62	67	25	77
Usable landed/entry mass ratio	0.70	0.67	0.64	0.66	0.64
Usable landed mass margin, percent	6.7	4.3	7.6	1.7	8.6

[a]After deorbit maneuver.

[b]Includes $(ERV)_1$ (see table XXI) and EEC or EOC (see table XXIII or XXIV).

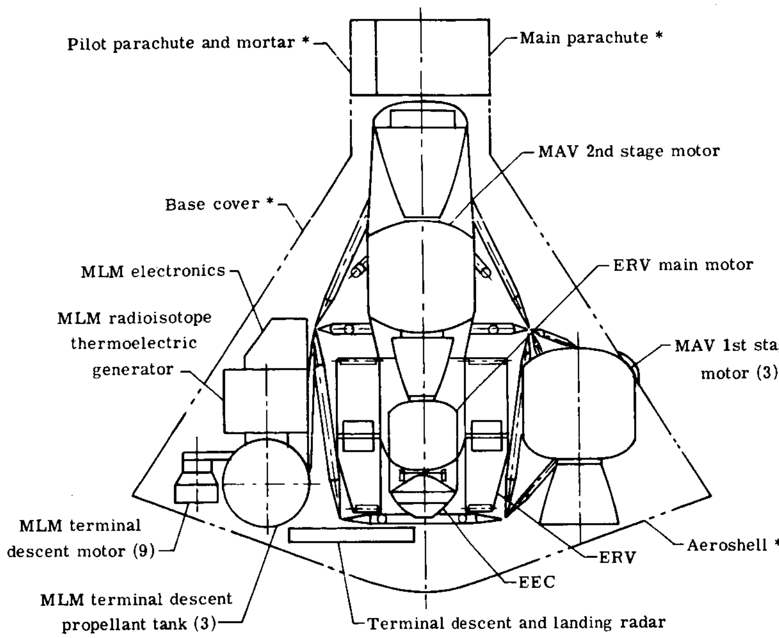


(a) Side view.

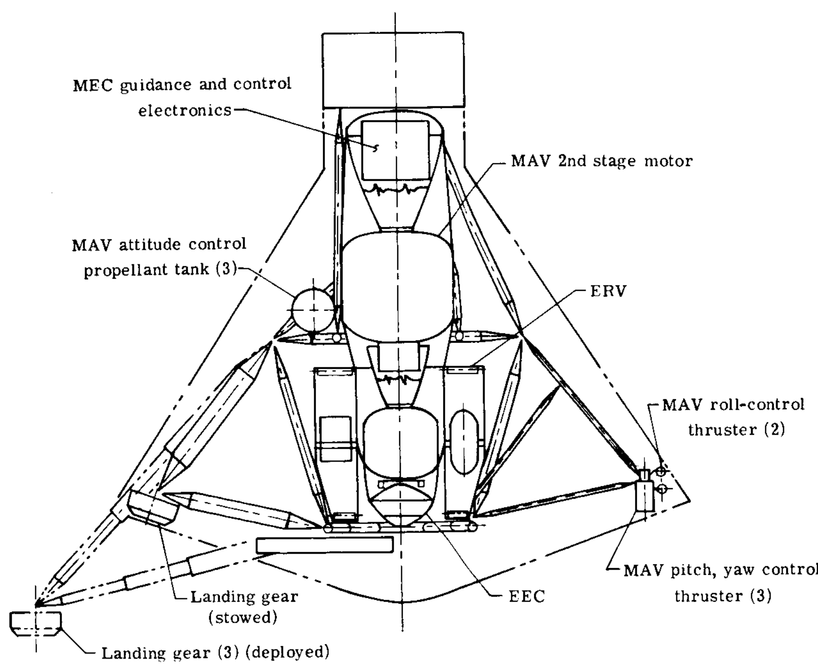


(b) Top view. Items marked with an asterisk are jettisoned during descent.

Figure 21.- Mars entry capsule baseline configuration and 3.51-m-diameter aeroshell.

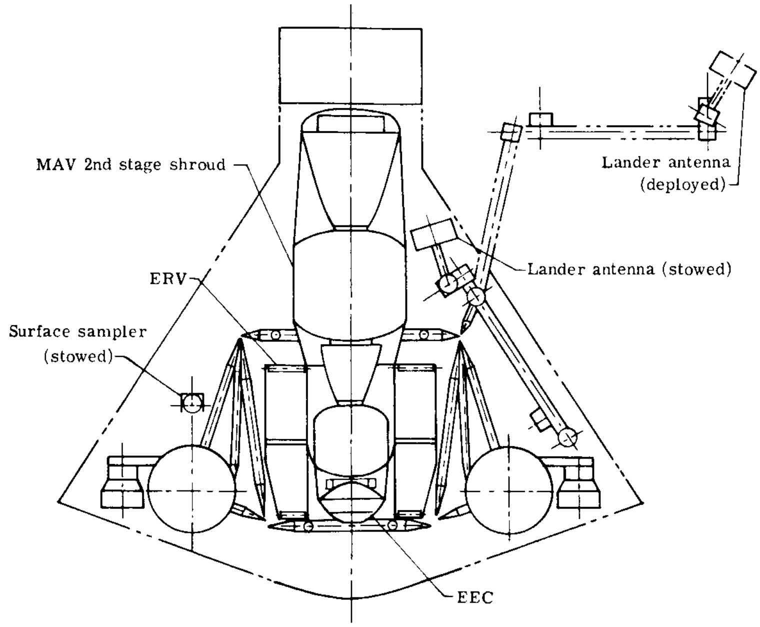


(c) Section A-A view. Items marked with an asterisk are jettisoned during descent.



(d) Section B-B view.

Figure 21.- Continued.



(e) Section C-C view.

Figure 21.- Concluded.

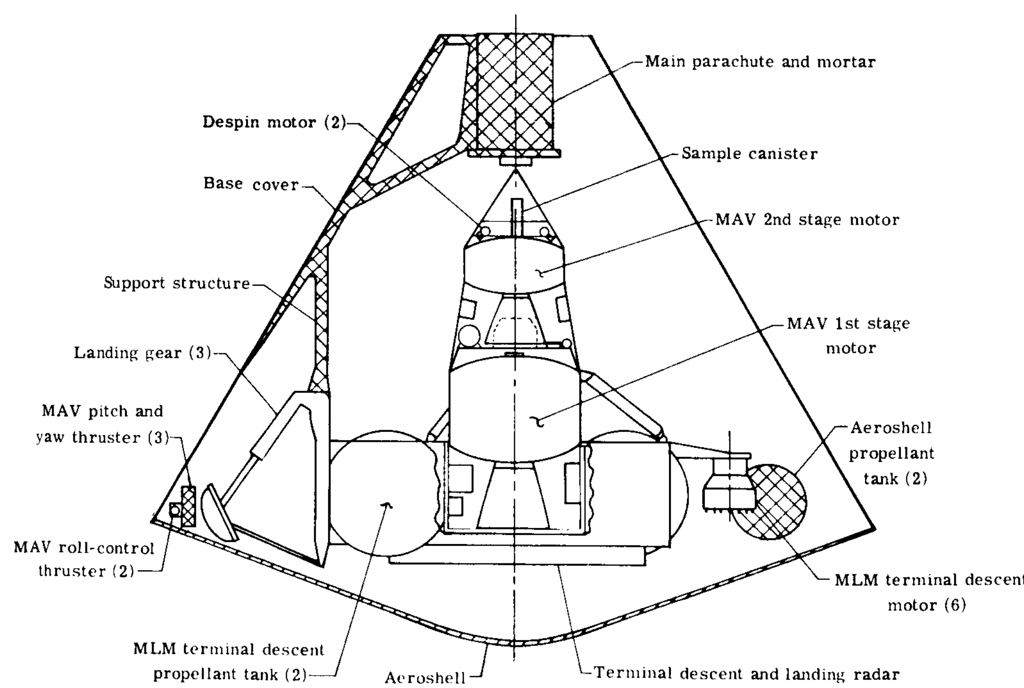


Figure 22.- Mars entry capsule with MOR spin-stabilized second-stage MAV and 3.51-m-diameter aeroshell.

Aeroshell

The aeroshell must provide a protective cover over the front of the entry capsule and produce a high aerodynamic drag to reduce the hypersonic entry speed to low supersonic in the lower atmosphere where a high performance parachute can be deployed. During this time the aeroshell attitude control system will maintain roll control to keep the lift vector properly oriented. At the end of its phase, it must separate cleanly and move out of the path of the MLM/MAV.

A summary of the aeroshell masses is presented in table XIV for each of the five concepts. A mass-scaling formula based on the Viking '75 aeroshell was used to establish the mass required for the proper structural and heat protection changes for each concept.

TABLE XIV.- SUMMARY OF SYSTEMS MASSES FOR MARS AEROSHELL

System	Mass, kg, for –				
	$(A/S)_1$	$(A/S)_2$	$(A/S)_3$	$(A/S)_4$	$(A/S)_5$
Structure	139	117	97	103	83
Thermal protection	66	58	51	53	43
Attitude control	63	54	44	144	100
Radar antenna	2	2	2	2	2
Power and wiring	5	5	5	5	5
Science	1	1	1	1	1
Total	276	237	200	308	234

Structure.- The Viking 140° total angle conical forebody, which is a strut- and ring-stiffened construction with an aluminum skin, has been used for all the MSSR aeroshells but with calculated increases in structural strength to withstand much higher dynamic-pressure loads. The aluminum design is protected from the extreme entry heating by a graduated layer of a phenolic nylon ablator. In addition, the internal components are protected by insulation blankets.

Attitude control.- The attitude-control subsystem for the MEC is located on the aeroshell. It operates from the time of MEC separation from the MCM or MOV until separation of the aeroshell from the lander. It consists of thrust nozzles, two spherical propellant supply tanks, and the appropriate valves and lines. The subsystem is controlled by the DHAC system located within the MAV. The attitude-control subsystem uses monopropellant purified hydrazine with blowdown gaseous helium for pressurization. Attitude-control impulse is provided by twelve enlarged Viking-type thrusters: four yaw, four pitch, and four roll. The pitch and yaw nozzles provide the deorbit retrothrust capability in the case for out-of-orbit entry and the MOR. The deorbit maneuver requires considerable additional propellant as well as tankage. An offset center of gravity is used during entry to produce the proper angle of attack; propellant is provided to damp the oscillations.

Telemetry.- The aeroshell has no telemetry system but uses a flat-plate antenna for the altimeter radar located on the MLM. This radar permits the sensing of altitude soon after the aeroshell begins entry providing data for the proper parachute deployment point and subsequent aeroshell separation.

Pyrotechnic devices.- Pyrotechnic devices to enable aeroshell separation are provided by the MLM.

Science.- During aeroshell entry, the only science measurements to be taken are stagnation-point temperature and pressure, providing for subsequent atmospheric reconstruction with altimeter data.

Parachute System

The parachute system which includes the parachute, mortar, truss, and base cover must provide continuing deceleration of the MLM/MAV following the aeroshell phase. Therefore, it must house and deploy a parachute which can take advantage of the tenuous Martian atmosphere. The parachute system also provides a conical base cover to protect the MAV, and a compartment at the top of the truss work to contain the pilot parachute, parachute mortar, and the main parachute. The parachute compartment is smaller for MEC_3 and MEC_5 (557 kg MAV) since the main parachute is deployed with a mortar. A summary of masses is presented in table XV for each of the two parachute system

TABLE XV.- SUMMARY OF MASSES FOR PARACHUTE SYSTEMS

System	Mass, kg, for –	
	$(P/S)_1$	$(P/S)_2$
Structure	66	61
Thermal protection	30	25
Parachutes	122	43
Mortars	5	13
Main parachute canister	6	
Pyrotechnics and wiring	3	3
Total	232	145

designs. The parachute masses were derived from a mass-scaling equation based on the Viking '75 parachute design.

Structure.- The structural design of the parachute system was based on the special MSSR requirements to house and deploy a large parachute from the top of a high-standing ascent vehicle. Since the MAV itself must not be penalized to provide this structural support, a truss structure surrounding the MAV was required. The two MAVs (baseline and MOR with three-axis control) use a strut structure. The parachute system for the MOR/MAV which is spin stabilized uses a milled aluminum structure.

Thermal protection.- The base cover, which protects the MAV, is aluminum sheet covered with a 0.8-cm layer of cork silicate for thermal protection. The high angle of attack of the MEC during entry will produce heating on the base cover. To protect the internal components, sheet insulation is used on the underside of the base cover. A bubble will be required in the base cover to accommodate the MAV propellant tanks. (See fig. 21.)

Parachutes.- The parachutes were based on the type of MAV being carried to the Mars surface. For the low-mass spinner MAVs, the Viking type and size of parachute was assumed. This is a disk-gap-band (DGB) type with a 16.1-m diameter and is deployed with the Viking mortar. The deployment conditions of $M_d \leqq 2.2$ and $q_d \leqq 431\ N/m^2$ were used, these values being the Viking values at the time of this analysis.

The other MECs required more decelerating capability and a larger DGB parachute. Deployment conditions of $M_d \leqq 2.3$ and $q_d \leqq 575\ N/m^2$ were used. To deploy this parachute, however, an excessively large mortar would have been required, and the reaction load for such a large mortar was inadmissible. Consequently, a mortar-deployed 6.2-m-diameter ribbon pilot parachute was selected to deploy the main parachute at the proper dynamic condition.

Pyrotechnics.- The parachute system will require pyrotechnics to cut the radioisotope thermoelectric generator coolant lines at the time of the parachute separation from the MLM. These lines provide external cooling of the RTG during launch pad operations. Secondly, pyrodevices will be required to release the end cap of the parachute compartment prior to mortaring out the parachute. The unlatching of the parachute system from the MLM is handled by pyrotechnic devices on the MLM itself.

Mars Lander Module

The MLM, the last element of the MEC involved in the entry and landing phase, must supply the retropropulsive capability to bring the MAV and surface sampling instrumentation safely to Mars surface. It must also provide a low mass launch platform for the MAV. A functional block diagram of the MLM is shown in figure 23.

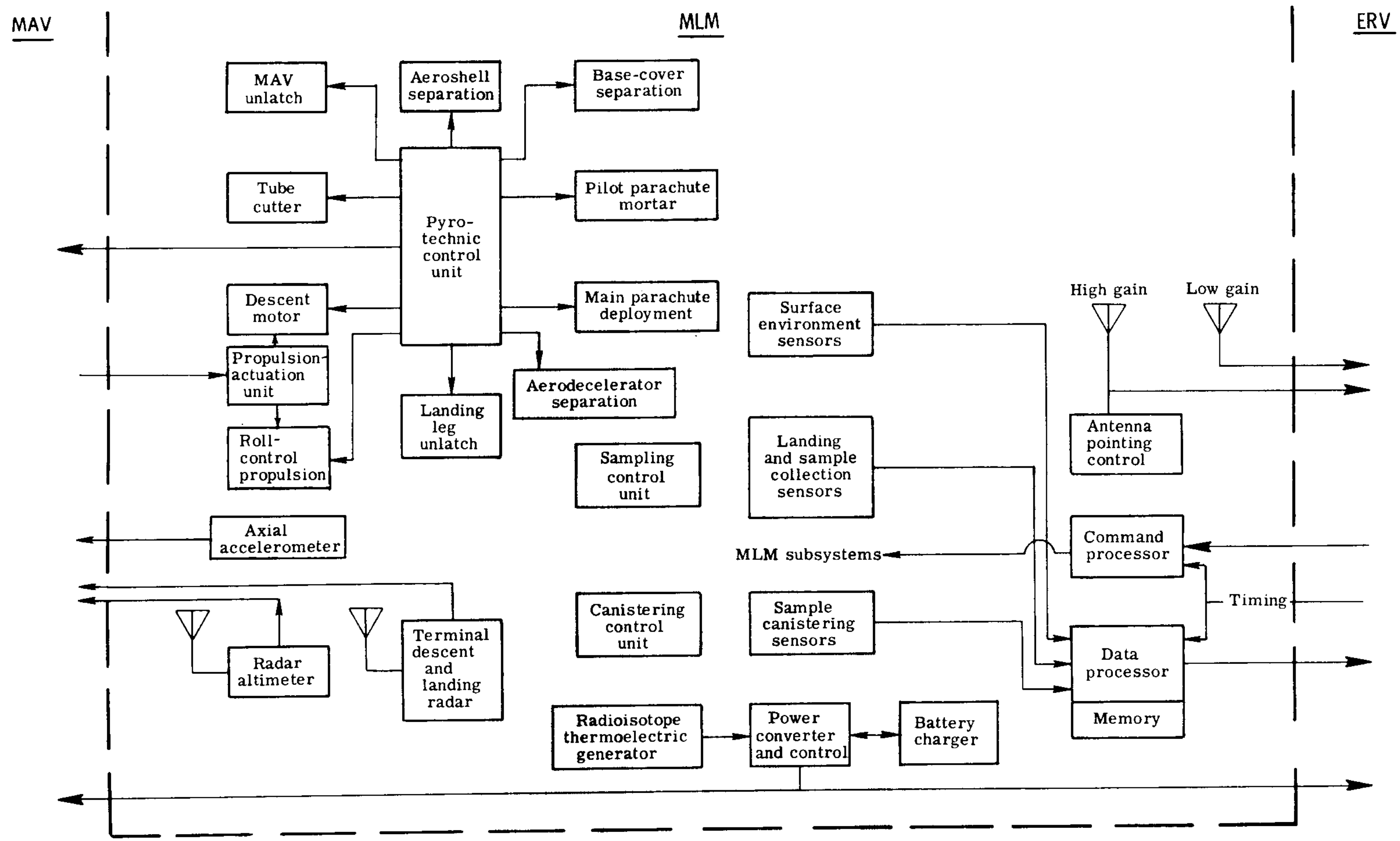


Figure 23.- Mars lander module functional block diagram.

The MSSR MLM's propulsion system, terminal descent and landing radar, and radar altimeter are of Viking design; however, it was necessary to make extensive changes from the Viking MLM structure and configuration to accommodate efficiently the MSSR need. The MSSR mission will use a minimum of surface science so that the MLM has been reduced to a light but sturdy landing frame.

The MLM consists of an integrating structural frame work and three landing legs; the terminal descent propellant subsystem; telecommunications equipment; the pyrotechnics control unit; the sampler and the equipment for processing and loading the sample into the sample capsule; a satellite DHAC unit to control the operations and process the data results; environmental monitoring instrumentation; and a radioisotope thermoelectric generator. Each of these elements and their variations is discussed individually. A summary of the MLM mass is shown in table XVI. A sketch of the deployed baseline MLM is shown in figure 24. This sketch is typical also for the three-axis MOR-type MLMs. Sketches of the baseline MEC shown in figure 21 illustrate many of the MLM components.

Structure.- The main structure for all the MLMs was a lower frame assembly on which three landing legs were fastened and above which the MAV was supported. The MAV was held above this base frame by an upper and lower frame intertruss. (The upper frame is part of the MAV and is separated prior to lift-off). The lower frame assembly, intertruss, and landing-gear masses varied according to the MAV being supported. The remaining structural elements which consisted of supports for peripheral instruments or equipments were the same for all MLMs from one MLM type to another.

Telecommunications.- The telecommunication system for the MLM provides for two operations: altitude sensing during descent, and Earth communication during surface operations. However, only the MLM bearing the MOR spinner type MAVs contained both the electronics and the antenna for Earth communication. The baseline MLM and the MLM bearing the three-axis MAV did not contain the electronics for Earth communication since the electronics were needed during ascent. The electronics were located in the ERV on the baseline mission and on the MAV for the three-axis MAV mission case. All MLMs have low-gain and high-gain antenna for radiating and receiving at S-band frequency. The high-gain antenna is a steerable design which must be deployed on landing.

Power.- The power subsystem for all MLMs consists of a Snap-19 radioisotope thermoelectric generator (RTG), converters, switching and control circuits, a battery charger, and an interconnect with a 280 W-hr battery located in the ERV (or MAV) to meet high level short demands. The RTG, which delivers 35 to 40 watts continuous power, supplies the nominal power during entry, landing, and launch operations.

TABLE XVI.- SUMMARY OF SYSTEMS MASSES FOR MARS LANDER MODULES

System	Mass, kg, for –				
	$(MLM)_1$	$(MLM)_2$	$(MLM)_3$	$(MLM)_4$	$(MLM)_5$
Structure	157.7	128.2	92.4	128.2	92.4
Telecommunications	37.1	37.1	43.8	37.1	43.8
Power	19.2	19.2	19.2	19.2	19.2
Data handling and command	1.3	1.3	1.3	1.3	1.3
Pyrotechnics	9.3	9.3	9.3	9.3	9.3
Thermal control	22.7	22.7	22.7	22.7	22.7
Mechanisms	11.6	11.6	11.6	11.6	11.6
Wiring	10.3	10.3	10.3	10.3	10.3
Propulsion and attitude control	353.9	214.2	152.4	255.0	156.9
Science	29.8	29.8	29.8	29.8	29.8
Total	652.9	483.7	392.8	524.5	397.3

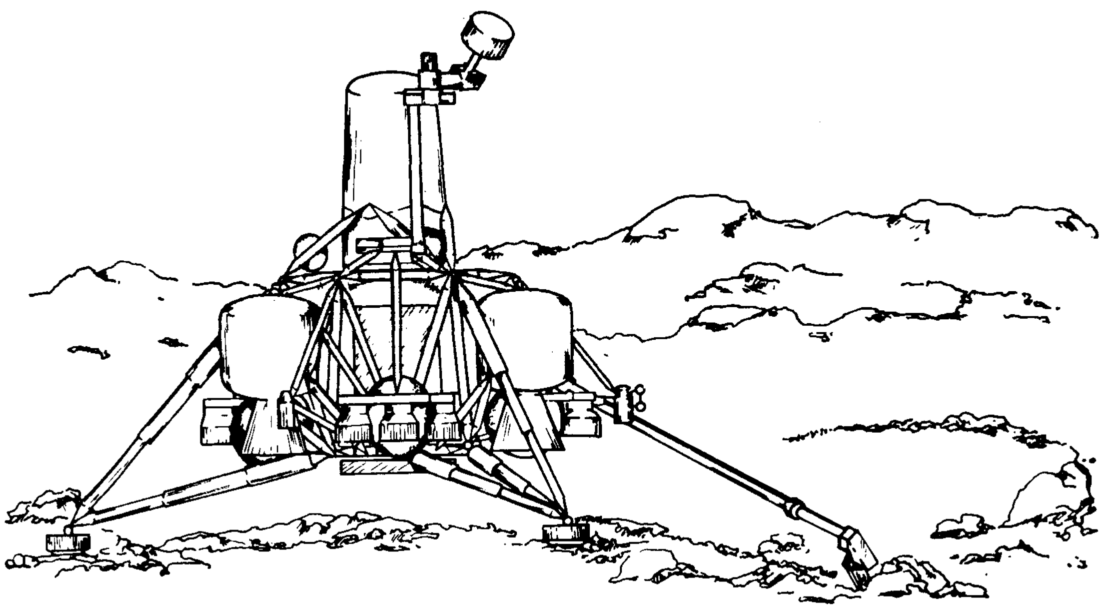


Figure 24.- Typical Mars landed configuration.

Attitude control.- Attitude control of the MLM during terminal descent is executed by the DHAC systems on the baseline ERV (or on the MAV for MOR missions); however, the axial accelerometer of the MLM and the inertial guidance of the ERV (MAV) provide the sensory inputs to the DHAC system. In addition, the terminal descent and landing radar gives guidance data for landing attitudes. The propulsive impulses required are supplied by the terminal descent propulsion subsystem. The propulsion subsystem will be discussed further.

Data handling and command.- The MLM data handling and command subsystem consists of a reduced version of the main DHAC system located in the MAV. The MLMs DHAC system operates as a satellite subsystem of that in the MAV. Surface operation commands received from Earth are processed by the DHAC system in the MAV (or in ERV), and then sent to the satellite DHAC system in code form to execute. The reverse of this applies to data. Extensive use of CMOS circuit elements has been made to minimize mass, volume, and power requirements.

Pyrotechnics.- The pyrotechnics subsystem of the MLM is the heart of the pyrotechnics activity for the entry, landing, and MAV launch. The pyro control unit and the outlying pyro devices are used for separations, ignitions, and other critical activities which must take place around the MLM. All MLMs have an identical requirement for pyro activity.

Thermal control.- Thermal control of MLM components consists essentially of providing a balance between heat output from the RTG and heat loss to the Martian environment. By using a thermal-transfer switch designed after that of Viking, this flow can be regulated. The RTG will require a wind shroud to prevent fluctuations in its heat output due to the high-velocity, variable Martian surface winds anticipated. Insulation of the electronic components will be needed to provide protective coverings and ten radioisotope heaters will be used to warm outlying instruments.

Mechanisms.- The MLM will require mechanisms for the high-gain antenna deployment and pointing (reported under telecommunications), and positive release mechanisms for the MAV release, the descent tank releases (fastened to MAV during entry and landing), and the landing-gear releases.

Terminal propulsion.- The terminal propulsion subsystem for the MEC is located on the MLM. It operates from parachute separation until touchdown is sensed by the landing-gear force sensors. This subsystem provides the roll control of the MLM/MAV, and pitch and yaw control during propulsive descent. Pitch and yaw control are provided by differential throttling of the three pods of main thrusters. Roll-control thrusting comes from nozzles located on each of the three terminal descent propellant tanks of the MLM. The roll thrusters required for the very massive baseline MLM/MAV are slightly larger. To supply the requirement for propellant of each type MLM, different sizings of tanks and plumbing were required. In all cases, however, three spherical tanks were used. For the out-of-orbit spinner-type-MLM, Viking tanks and plumbing could be used, whereas all others required enlargements. The Viking descent motors (2850-N nominal thrust) were used exclusively; nine motors were required for the baseline MLM whereas six motors were sufficient for the MOR MLMs. These motors use a special 18-nozzle diffuser design to minimize alteration of the landing site at touchdown.

The propellant used is purified hydrazine (monopropellant) with helium gas used for pressurization of the propellant. The pressurization gas is contained in the propellant tanks separated by a bladder. Roll thrusters of the terminal descent propulsion subsystem are fed from this same propellant supply. The nominal I_{sp} for this propellant with the 18-nozzle design of the descent motors is 2098 N-sec/kg.

Science.- The science system will acquire, process, and store the sample; it will provide data on the sample, the sampling equipment, and the environment during sample acquisition; and it will continuously monitor the surface environment.

The sample acquisition and storage equipment generally meets the recommendations discussed in the section "Science Requirements," and the sample acquisition operations are discussed in the section "Surface Operations." The science instruments will provide the measurements recommended in the section "Science Requirements." These instruments include lander leg force sensors for measuring landing impact forces, temperature sensors, sampler position sensor, sample scoop load sensor, ambient light photometer, humidity sensor, and wind-velocity sensor.

DESCRIPTION OF ENTRY PROFILES

Entry profiles were generated for each of the five entry vehicle designs. A sequence of events for the baseline mission is shown schematically in figure 25.

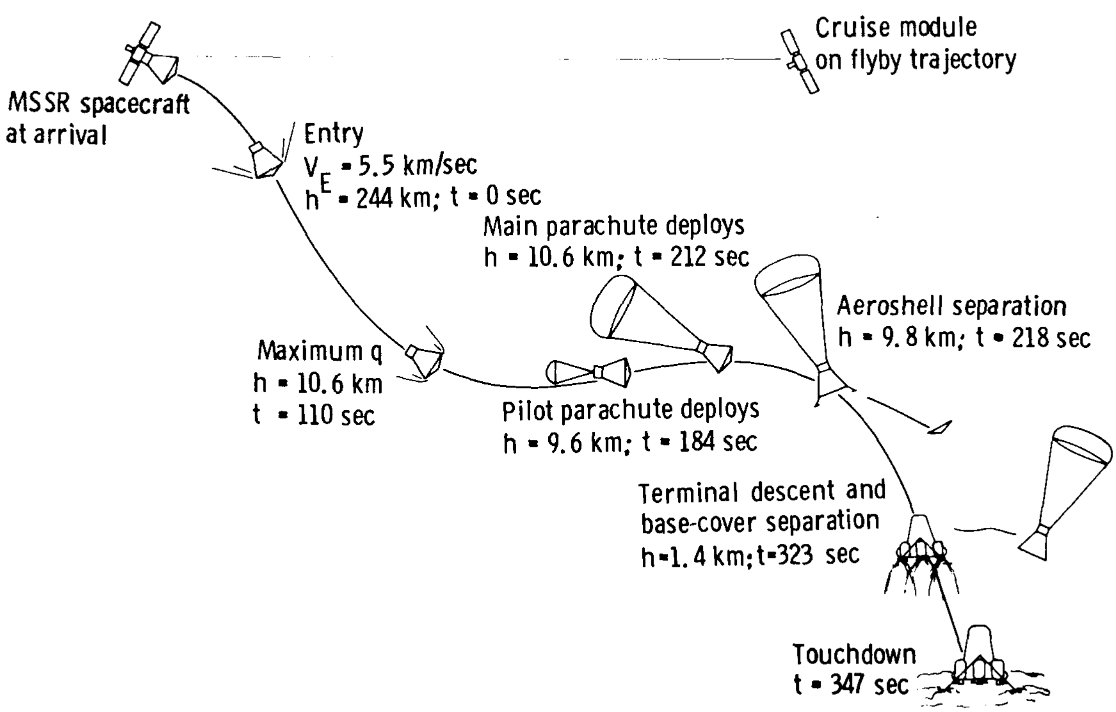


Figure 25.- Baseline mission Mars entry events.

In general, the Mars entry and landing phase will be accomplished in three stages of deceleration in the manner of Viking, beginning at an altitude of approximately 244 km above Mars mean surface level; however, attitude-control maneuvers by the MEC must begin before entry. In the case of out-of-orbit entries, the MEC must also perform deorbit thrusting prior to entry.

All the MECs hold a Sun-referenced attitude until approximately 6 minutes before reaching an altitude of 244 km above Mars. At this time the DHAC system commands a reorientation to an angle of attack prescribed for entry of the respective MEC design.

Selected Entry Parameters

The MSSR entry profiles differed markedly from that of Viking because of the greater MSSR entry mass and higher L/D and entry angles used. The final entry parameters are summarized in table XVII. The $(MEC)_3$ and $(MEC)_5$ which carried the 557-kg MAV required the least change from Viking's entry parameters; its entry mass is increased by about 46 percent. Accordingly, the Viking-type parachute deployment scheme and deployment Mach number and dynamic pressure were used with the $(MEC)_3$ and $(MEC)_5$ which carry the 557-kg MAV. The $(MEC)_2$ and $(MEC)_4$ carrying the 1071-kg MAV used the base-line concept parachute and the deployment scheme involving the pilot parachute.

Entry Trajectories

Allowable entry trajectories for each of the five MECs are represented by a family of curves which fall within the respective entry corridors. Final entry data for each MEC are presented in figures 26 and 27. Figure 26 shows entry velocity with altitude for the five MECs. The variation of altitude, Mach number, and q with time from entry to touchdown is illustrated in figure 27 for each of the five entries.

SUMMARY OF ENTRY AND LANDING SYSTEM ANALYSIS

It was found possible to land sufficient mass on Mars, by using one Titan III E/Centaur as the Earth launch vehicle, to support the baseline mission (and all but one of the MOR missions). Based on the analysis herein, this capability existed without resorting to the use of increased aeroshell diameters. The use of the Shuttle/Centaur with increased diameters up to 4.27 m would provide significant increases in entry mass. A working ratio between usable landed mass and entry mass was found to be about two-thirds for the entry masses and systems employed. The more massive entry vehicles proved to be the more efficient in providing usable landed mass. Enlarged Viking-type parachutes using increased M_d and q_d values appeared to be reasonable for obtaining the increased parachute performance needed for MSSR payloads. The Viking terminal

TABLE XVII.- ENTRY AND LANDING CHARACTERISTICS FOR THE SELECTED MARS ENTRY CAPSULES

Parameter	Values for Mars entry capsules				
	$(MEC)_1$	$(MEC)_2$	$(MEC)_3$	$(MEC)_4$	$(MEC)_5$
Atmosphere model	Mean	Mean	Mean	Mean	Mean
Entry velocity, km/sec . . .	5.5	5.5	5.5	4.6	4.6
Entry mass, kg	2858	2086	1361	2086	1361
L/D (hypersonic)	0.32	0.26	0.18	0.28	0.18
Entry angle corridor, deg . .	-26.8 to -24.8	-24 to -22	-22 to -20	-22 to -20	-20 to -18
M_E/C_DA, kg/m^2	205	144	90	145	90
Maximum entry q, kN/m^2	40.8	25.9	14.4	17.7	9.0
Maximum total heat load Q_c, MJ/m^2	32.1	27.2	22.5	20.1	16.4
Maximum entry $\dot{q}_C$, kW/m^2	1248	1010	773	569	422
Maximum entry loading, Earth g units	19.5	17.4	16.5	12.1	10.5
Parachutes used	Pilot (6.2 m) plus main (22.9 m)	Pilot (6.2 m) plus main (22.9 m)	Viking (16.1 m)	Pilot (6.2 m) plus main (22.9 m)	Viking (16.1 m)
Pilot parachute M_d	≈3	≈3	-------	≈3	-------
Main parachute M_d	≦2.3	≦2.3	≦2.2	≦2.3	≦2.2
Main parachute q_d, N/m^2	≦575	≦575	≦431	≦575	≦431
Number of Viking descent engines used	9	6	6	6	6
Terminal descent altitude required, m	1400	990	792	1310	810

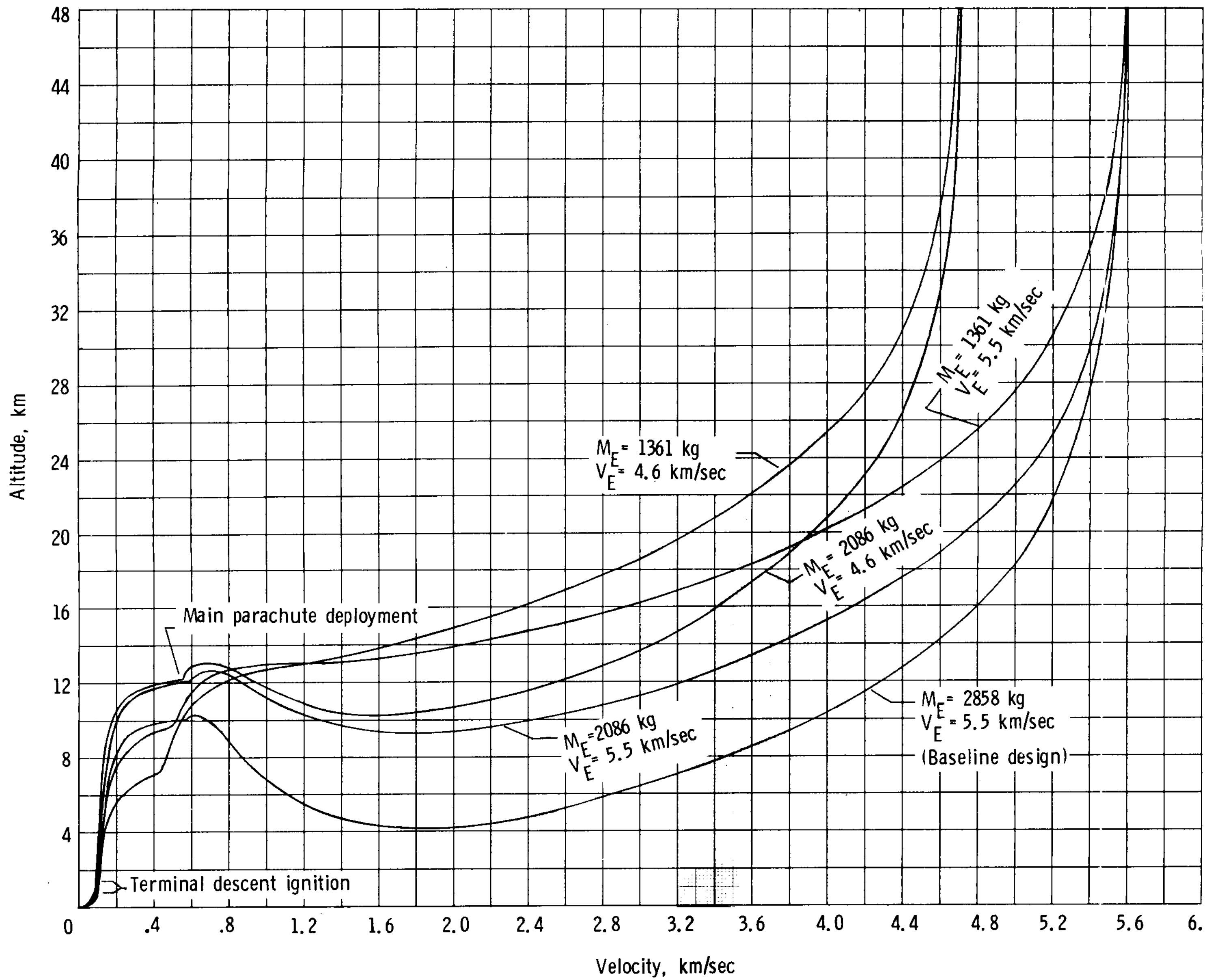
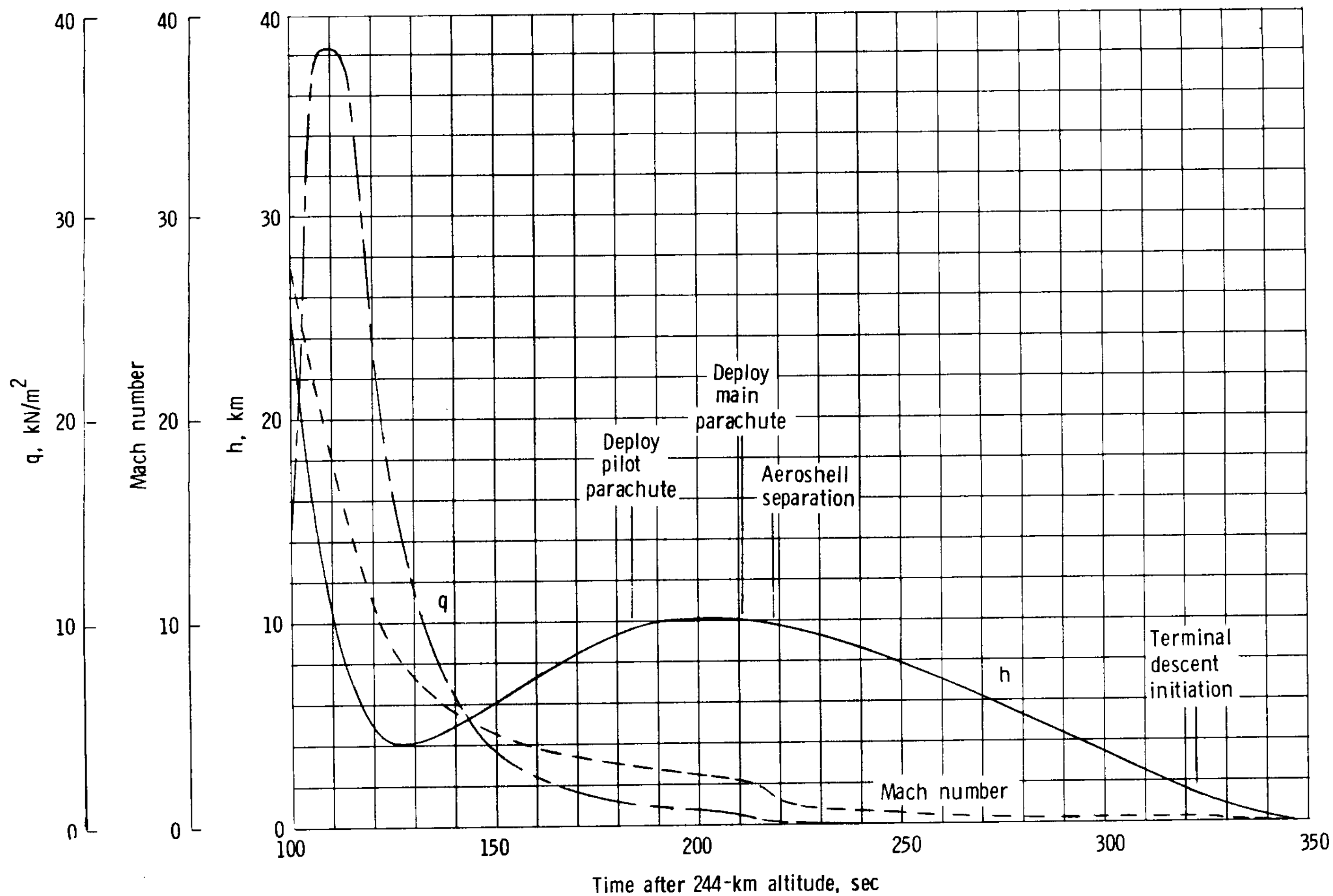
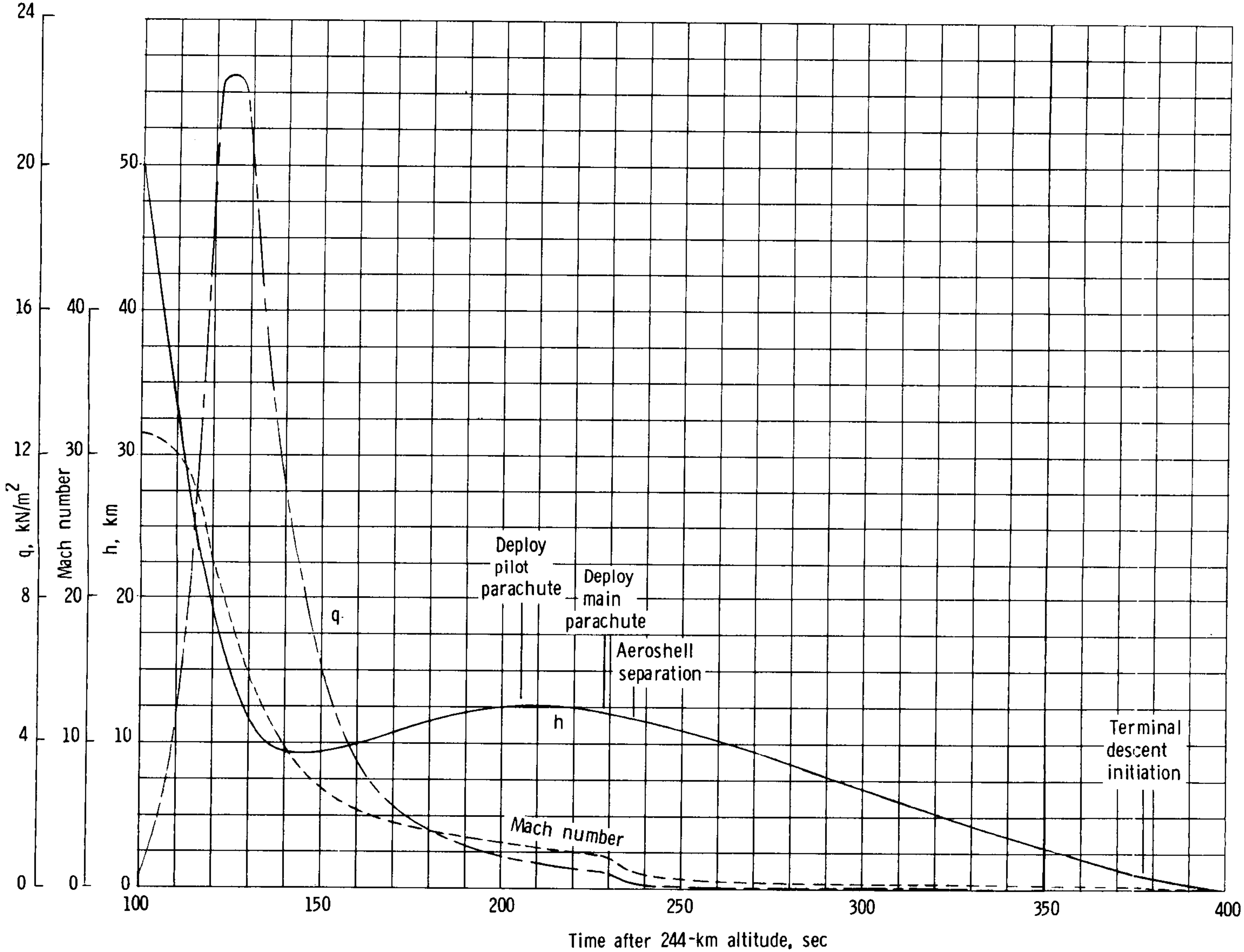


Figure 26.- Variation of Mars entry velocity with altitude for the five MEC designs by using the mean atmospheric model (see table XVII).



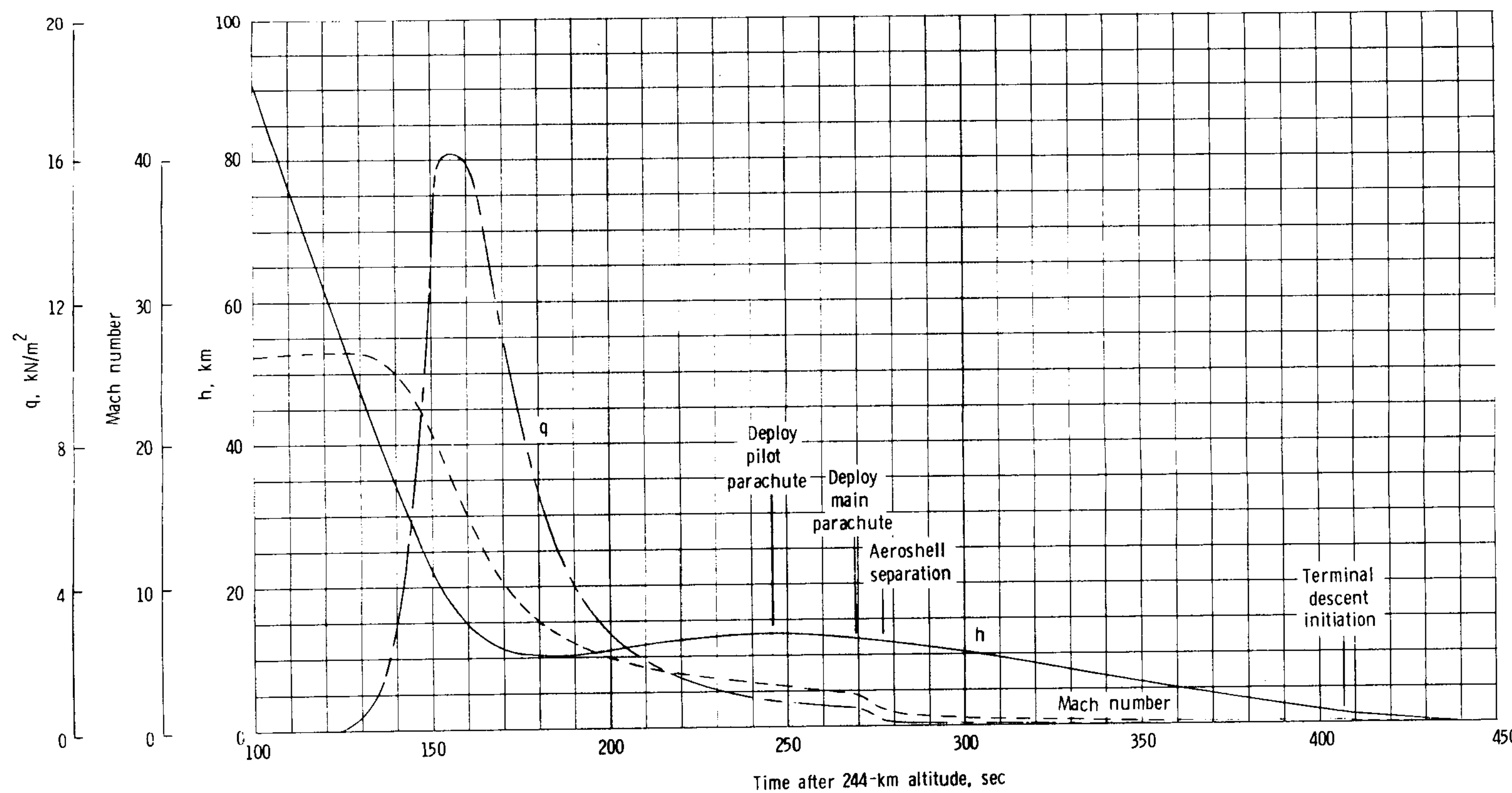
(a) Baseline mission.

Figure 27.- Variation of h, Mach number, and q with time during Mars entry and landing (see table XVII).



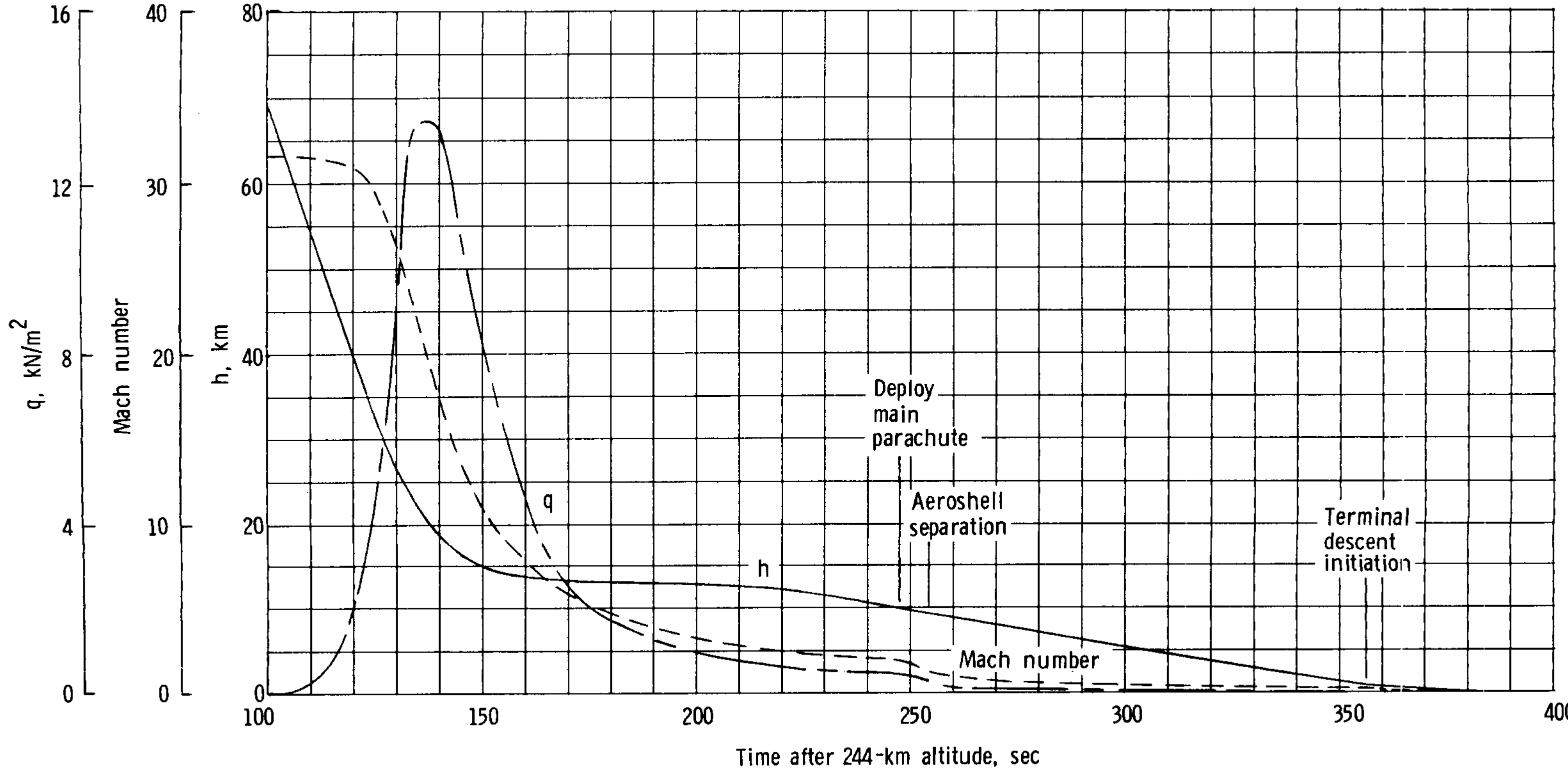
(b) Direct Mars entry/MOR (three-axis MAV) mission.

Figure 27.- Continued.



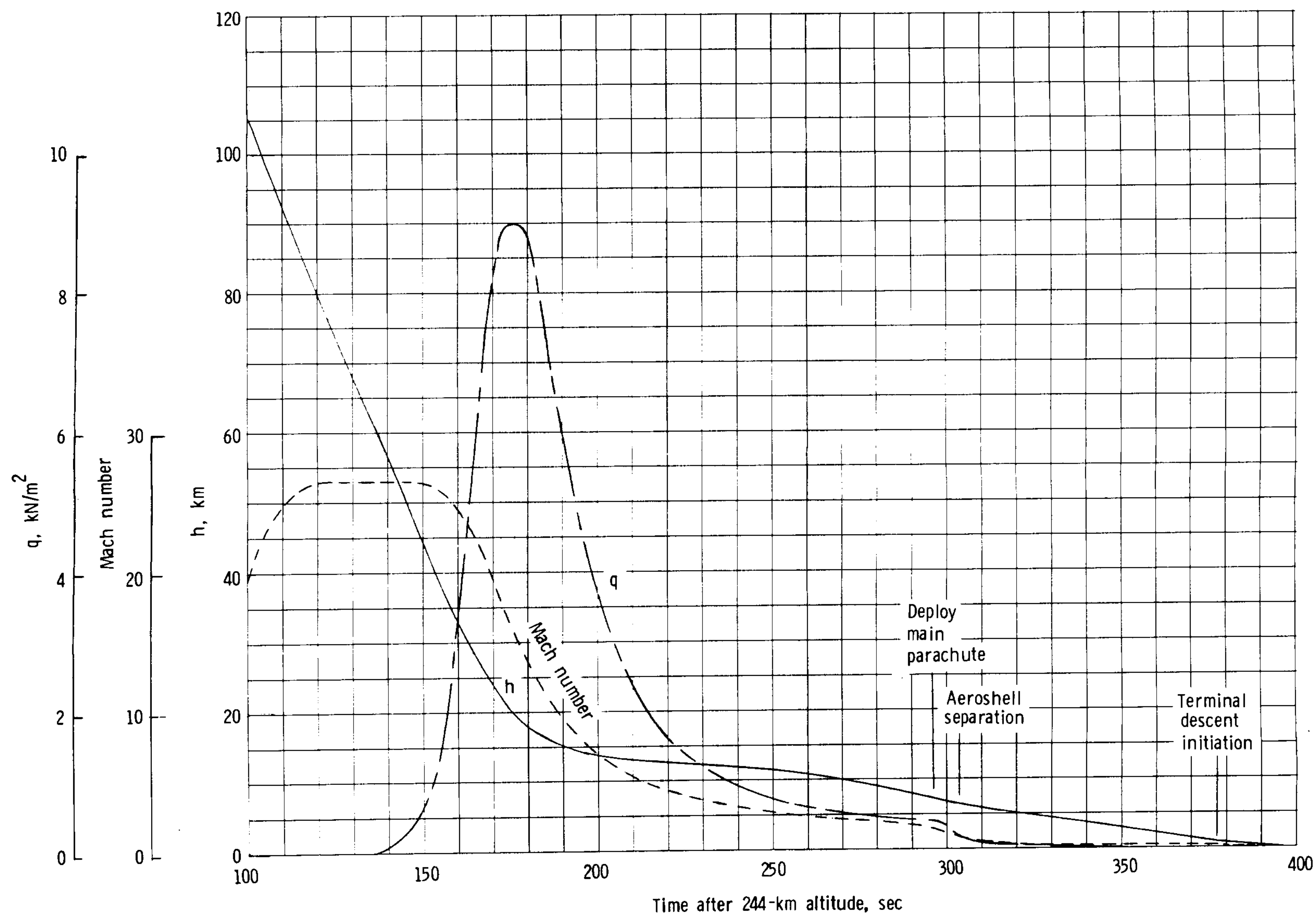
(c) Out-of-orbit Mars entry/MOR (three-axis MAV) mission.

Figure 27.- Continued.



(d) Direct Mars entry/MOR (spinner MAV) mission.

Figure 27.- Continued.



(e) Out-of-orbit Mars entry/MOR (spinner MAV) mission.

Figure 27.- Concluded.

descent propulsion system was found to provide sufficient thrust for the MSSR landings by increasing the number of engines used from 3 to 6 or 9 depending on the particular mission type.

SURFACE OPERATIONS

Mars surface operations consist of monitoring the surface environment, acquiring the sample, processing it, filling and capping the inner sample cup, transporting the inner canister to the outer cup location, and closing and sealing the outer cup before launch.

The outer cup is located in either the Earth entry capsule (EEC) or the Earth orbit capsule (EOC) for the baseline mission. (See fig. 21(c).) The outer cup is located in approximately the same position (at the front of the MAV second stage) for the MOR mission employing a three-axis controlled second-stage MAV. Its location is shown in figure 22 for the MOR mission employing a MAV with a spin-stabilized second stage.

A two-way communication link with the lander will allow Earth-based personnel to initiate the sample acquisition and handling sequence program and later to receive stored data which document the completed steps of the sequence and the results of the surface environment measurements. Since the sampling and sample-handling sequence will take place at night, communication to or from the lander during this time will not be possible. (The landing site will be on the backside of Mars during nighttime for a conjunction class mission.) Emphasis must be placed on making the sequence as automatic as possible and of minimum duration. The sooner the sample can be lifted off and placed in orbit, the less it would be exposed to a possibly deleterious environment (steep temperature gradients, dust storms, etc.). Hence, the acquisition sequence would be initiated as soon as possible after landing, but at a time and under the conditions meeting the specified science requirements.

Cursory study efforts were undertaken to examine and develop concepts for handling and canistering the Martian sample after it was processed. The Viking '75 surface sampler scoop, sampler boom, control assembly, and processor designs were selected for the sample operations up through processing. The Viking '75 designs are described in reference 13. The selected sample-canistering design and the sequence of surface operations are described.

Initially, the lander transmits to Earth ambient temperature, pressure, humidity, light intensity, wind velocity, data from which lander attitude can be determined, and any other data required to verify readiness of the sample-acquisition sequence. After verification, the command is sent which will start the sample acquisition sequence when night conditions reoccur. The command may include updating of the data handling and command (DHAC) program. Once initiated, the sequence is controlled by the satellite DHAC system onboard the Mars lander module (MLM). Built-in "holds" can be programed in the sample-acquisition sequence in the event of a malfunction, but such holds will be invoked only for a critical condition since they would substantially increase the duration of surface operations and because they could expose the sample to undesirable environments.

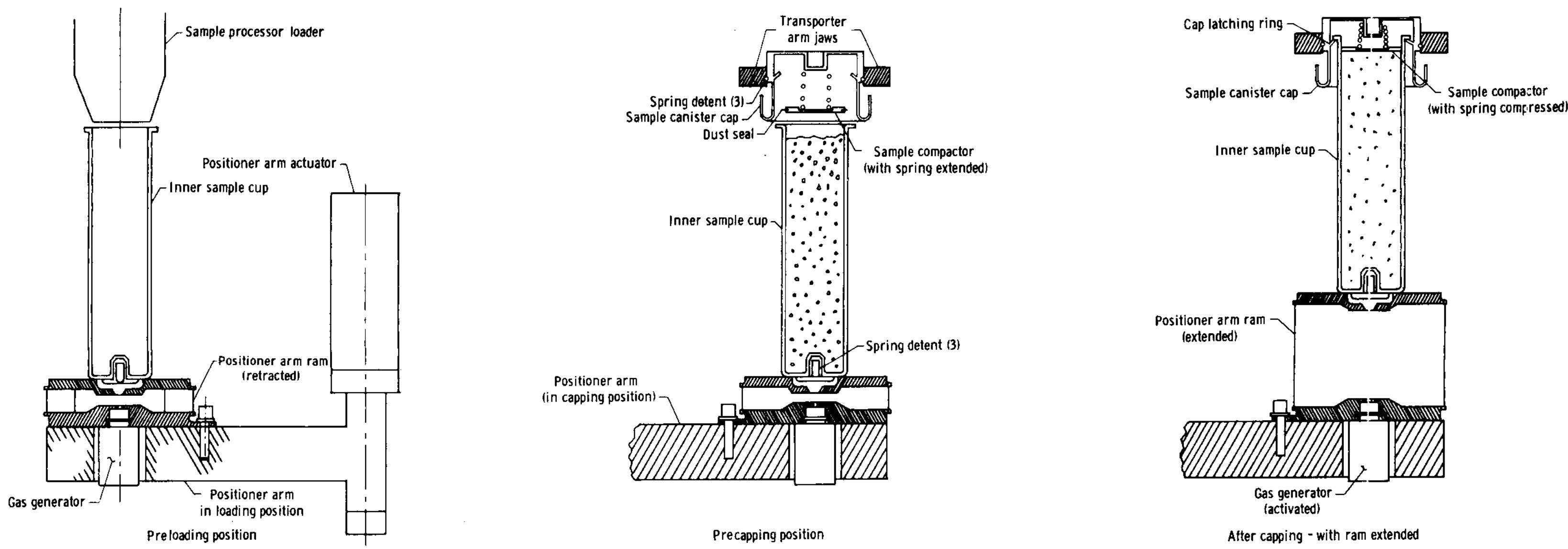
The sample acquisition sequence begins with the unlatching of the sample acquisition boom. The scoop cover is ejected, the boom positioned to the appropriate azimuth, extended to the desired length (approximately 3 m), and finally lowered to the surface (a ground contact limit switch stops travel). The MLM is now configured as illustrated in figure 24.

The acquisition and processing of the surface sample is similar to that used by the Viking '75 lander. The digging sequence consists of opening the scoop and extending the boom to drive the scoop into the surface material, closing the scoop, and then raising the boom. During these operations, data are stored for later transmission regarding boom azimuth and elevation position, boom extension position, boom loading force acting on surface, scoop head position, and scoop temperature.

Next, the scoop is positioned over the Viking-type processor and loader assembly which is equipped with a "full" sensor for closed-loop shutdown of the loading operation. If less than the full amount is processed the first time, the digging and loading cycle is repeated. The processor internal temperature is monitored. The processor can sieve or grind the sample, as required, to obtain the particulate mix desired.

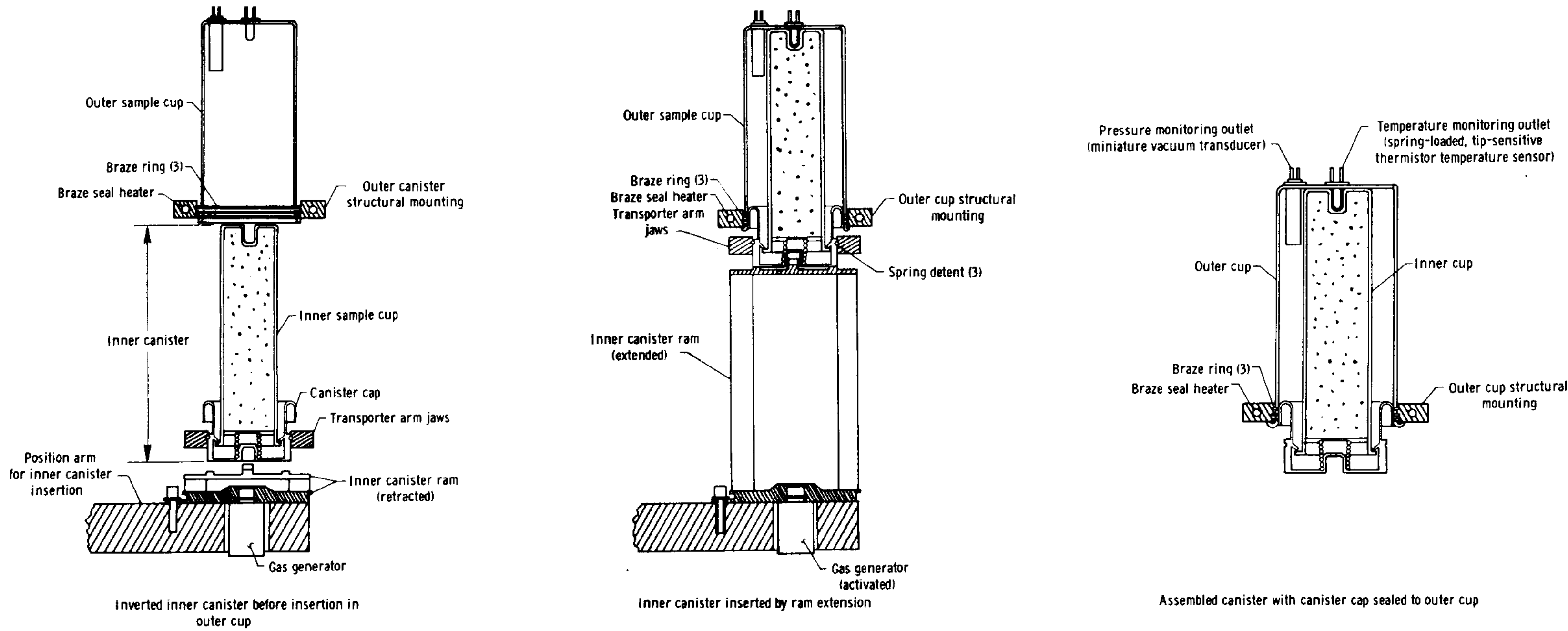
After the appropriate quantity of sample has been processed (200 grams is baseline) through the processor and loader, it is dumped into the inner sample cup. Figure 28 illustrates the elements involved in the sample loading sequence. In figure 28(a), the inner sample cup is attached by a small ram to a positioner arm. After inner cup loading the arm actuator swings the cup to a new position under the sample canister cap. The ram is forced upward by an expanding bellows arrangement driven by pyrotechnically activating a gas-generating material. The sample canister cap contains a spring-loaded sample compactor with a wiper lip that forms a dust seal for the inner cup.

As the ram drives the cup and cap together, the ring on the inside of the cap clips over the lip of the cup and locks them together mechanically. Sensors will indicate the completion of each operation in the entire sample handling and loading sequence for the purposes of later transmission to Earth-based personnel as well as to enable the next operational step in the programed sequence to be initiated. The assembled inner canister, now held firmly by the jaws of the transporter, is pulled off the ram spring detents and transported to a new inverted position under the outer sample cup. (See fig. 28(b).) Here, a second arm with an extendable ram swings into place and the inner canister is pressed up into the outer cup.



(a) Loading and capping of inner sample cup.

Figure 28.- Sample loading sequence.



(b) Enclosure and sealing of sample by outer cup.

Figure 28.- Concluded.

The final steps are to activate the braze welding rings which seal the inner canister to the outer cup and to retract the ram. For the baseline missions the outer cup is part of the EEC (or EOC) which is seen in figure 21(c). For the MOR missions the outer cup is in the bottom of the MAV and must be carried to the EEC (or EOC) in orbit. There it is fixed in place by a second set of braze weld rings.

The final insertion and sealing operation instruments the canister for monitoring of the sample by telemetry. Sample temperature variations should be sufficiently slow so that temperature measurement as in figure 28(b) will provide representative sample temperature. The outer canister can contain a low-pressure transducer to detect leaks through the outside of the sealed canister assembly.

ANALYSES, DESIGN, AND OPERATION OF MARS ASCENT SYSTEMS

Within a few days after a sample has been acquired and all functions on the surface performed, the sample will be launched into a Mars orbit along with other support systems required to continue the mission. For the baseline missions these systems include those for station keeping, Mars departure propulsion, Mars-to-Earth cruise, and those required at Earth arrival. For the Mars orbit rendezvous (MOR) missions, the systems include those required to rendezvous and dock with the Mars orbit vehicle (MOV) and to transfer the sample to the Earth return vehicle (ERV). The vehicle which provides the propulsion to perform this phase is the Mars ascent vehicle (MAV). Studies were performed to identify optimum vehicle propulsion parameters and trajectory techniques for efficient Mars surface-to-orbit launches, and a rocket motor design study was carried out.

VEHICLE PARAMETER AND TRAJECTORY OPTIMIZATION STUDY

A study was performed to determine propulsion system characteristics and design parameters for an efficient (low-velocity loss) MAV launched on an optimized Mars surface-to-orbit trajectory. The study considered (1) single-stage, continuous-burn vehicles, (2) single-stage, dual-burn vehicles, and (3) two-stage vehicles. Some of the results of the study are published in reference 14.

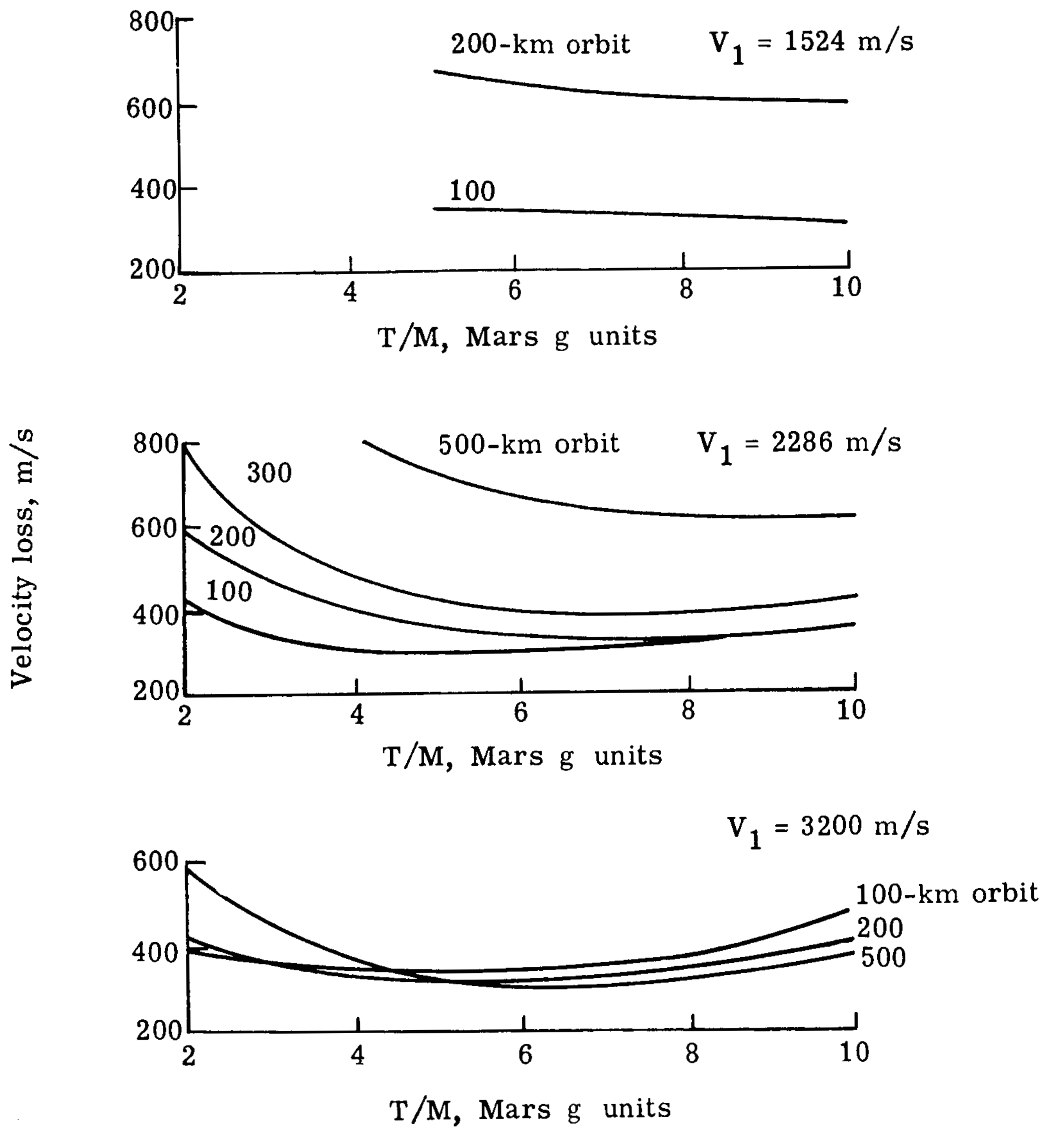
The trajectory optimization program of reference 15 was used. In this program initial flight-path angle and coast time between engine burn phases are adjusted to produce maximum in-orbit mass. It employs a spherical rotating planet and permits the input of an atmosphere which varies with altitude. The Viking mean density and maximum surface density atmosphere models were used. Plots of these density profiles are presented in figure 5. For the trajectory cases using the mean density profile, a constant value of $C_DA = 0.0929 \text{ m}^2$ was employed, and for the cases using the maximum density profile, a constant value of $C_DA = 0.1858\ m^2$ was employed. These cases are called the mean drag and maximum drag cases, respectively.

One measure of a surface-to-orbit vehicle's performance is velocity loss – the difference between ideal velocity of the vehicle and the minimum velocity required to achieve orbit. For the minimum-velocity case, the vehicle is launched from the equator into a posigrade equatorial orbit. Two Hohmann transfers are employed and there is no atmosphere. Minimum velocities are listed below for several Mars circular orbits:

Orbit altitude, km	Minimum velocity, m/s
100	3354
200	3403
300	3450
500	3538

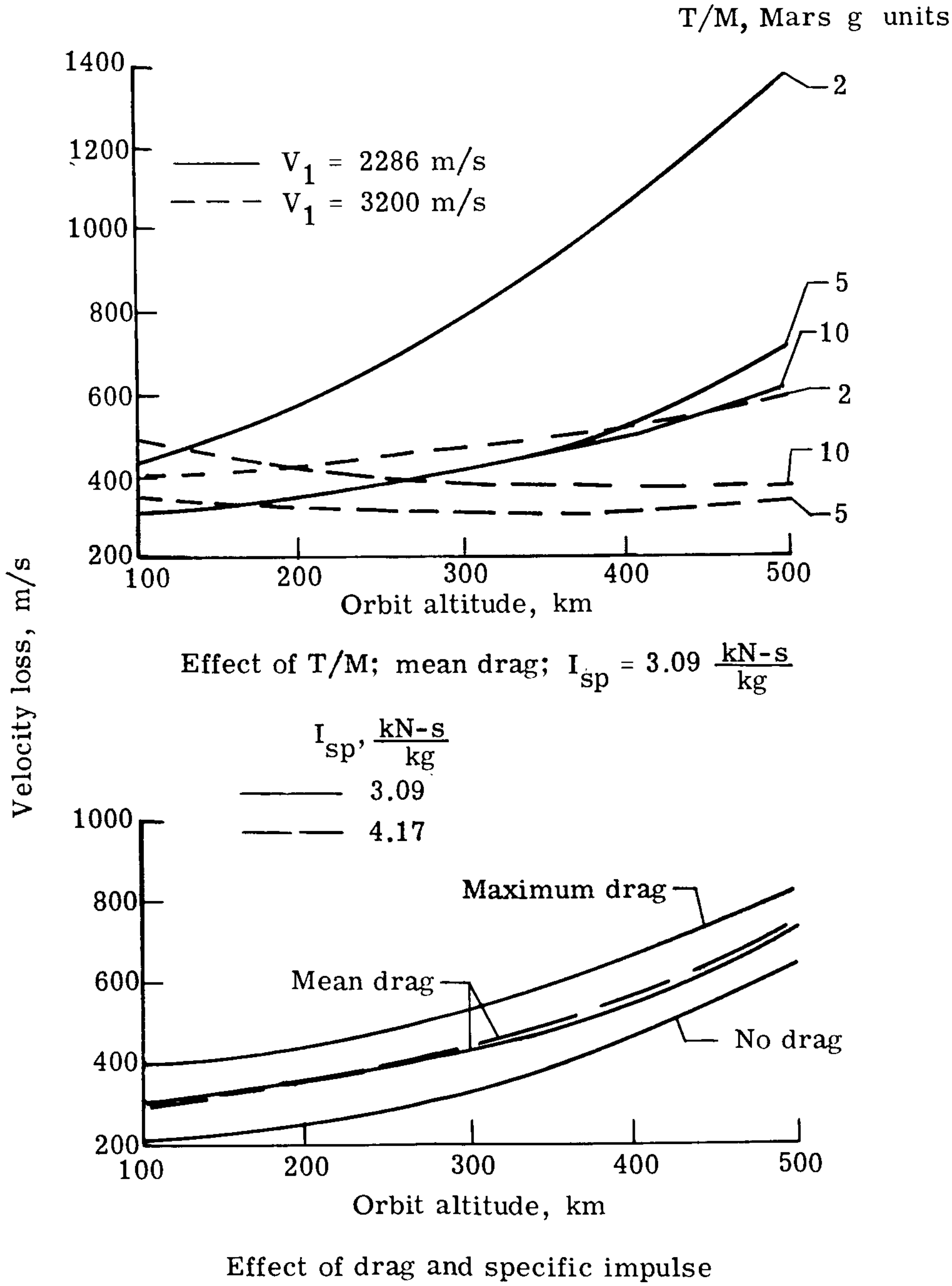
The results of reference 16 had indicated that single-stage, dual-burn Mars surface-to-orbit vehicles sustain significantly lower velocity losses than single-stage, continuous-burn vehicles and the current study verified that conclusion. Figure 29 presents velocity-loss data for single-stage, dual-burn vehicles launched to circular orbits at Mars. In figure 29(a) the values of T/M which give minimum velocity losses are seen to decrease with increasing values of V_1. Figure 29(b) shows the effect of orbit altitude on velocity loss for vehicles with different propulsion characteristics and drag characteristics. At the lower orbit altitudes, velocity losses are slightly less for vehicles with $V_1 = 2286$ m/s, but losses for the vehicles with $V_1 = 3200$ m/s and $T/M = 5$ and 10 are relatively insensitive to orbit altitude from 200 to 500 km. Losses due to drag are nearly constant with altitude and the effect of specific impulse is very small.

Table XVIII shows the altitude at second ignition, total velocity loss, and velocity loss for the second-burn phase for some typical single-stage, dual-burn vehicles and two-stage vehicles launched to circular orbits. Single-stage dual-burn vehicles and two-stage vehicles are seen to have about the same velocity losses, and essentially all losses are seen to occur during the first-burn phase. The trajectory-parameter plot for a typical single-stage, dual-burn vehicle (fig. 30) launched to a 100-km circular orbit illustrates the reason for these similarities between single-stage dual-burn vehicles and two-stage vehicles. It can be seen that when the vehicle reaches the altitude of the required orbit, the thrust angle is approximately zero and thus the second-burn phase essentially provides horizontal thrust for orbit insertion. This characteristic produces two significant results: (1) the results of the study for circular orbits are directly applicable to the prediction of velocity losses for injection into elliptical orbits; and (2) the guidance and



(a) Variation with T/M; mean drag; $I_{sp} = 3.09 \frac{kN\text{-}s}{kg}$.

Figure 29.- Velocity-loss data for single-stage, dual-burn MAVs launched to circular orbits.



(b) Variation with orbit altitude.

Figure 29.- Concluded.

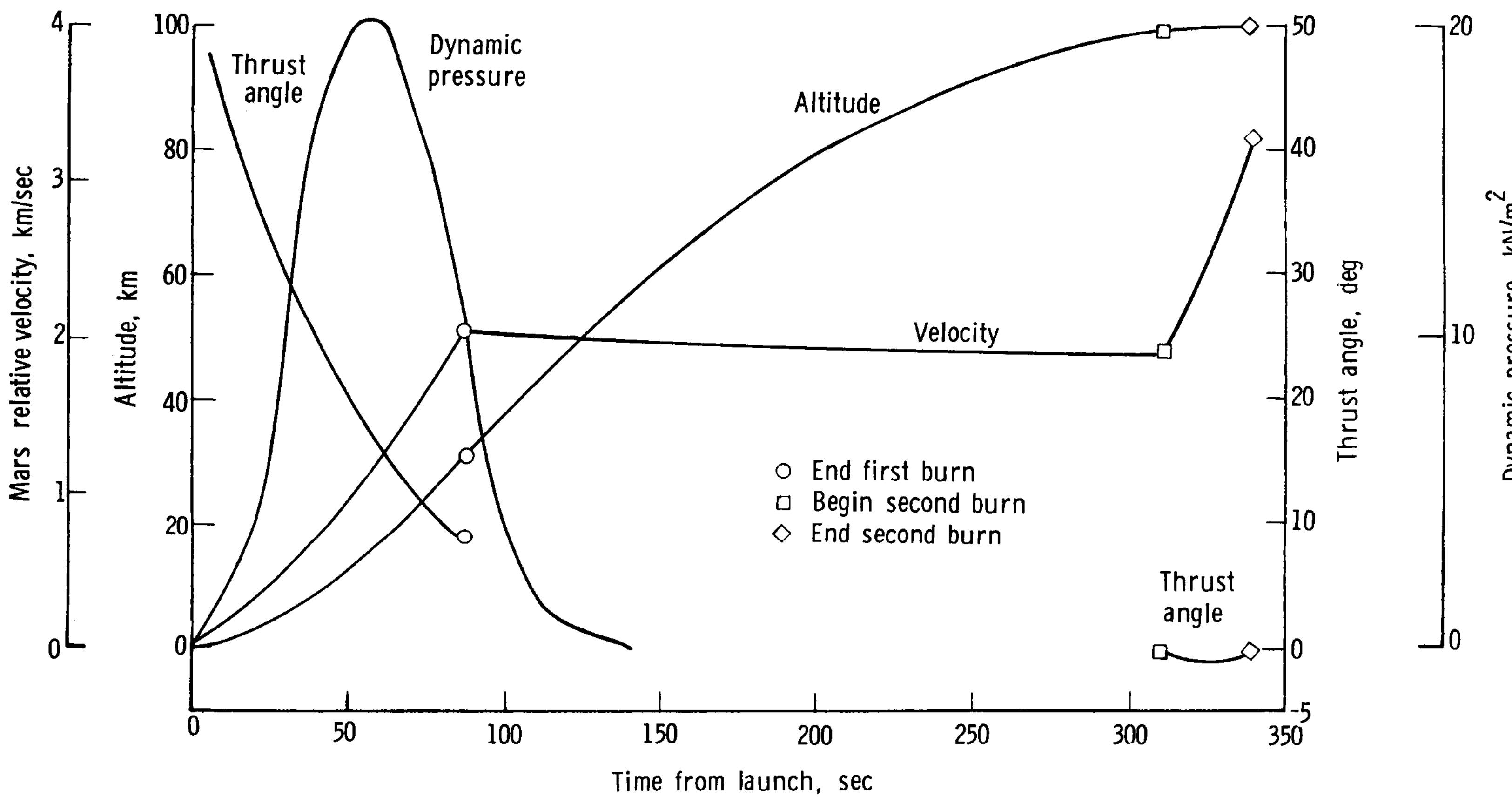


Figure 30.- Trajectory data for single-stage dual-burn MAV launched to 100-km circular orbit. V_1 = 2286 m/s; T/M = 5 Mars g units; $I_{sp} = 3.09 \frac{kN\text{-}s}{kg}$; mean drag.

TABLE XVIII.- ALTITUDE AT SECOND IGNITION AND VELOCITY LOSSES FOR TWO-BURN VEHICLES LAUNCHED TO CIRCULAR ORBITS

[DBV, single-stage, dual-burn vehicle; TSV, two-stage vehicle]

Orbit altitude, km	Altitude, km, at second ignition for –		Total velocity loss, m/s, for –		Second-burn velocity loss, m/s, for –		T/M, Mars g units
	DBV	TSV	DBV	TSV	DBV	TSV	
100	99.3	99.6	303	296	2	2	5
200	199.6	196.9	349	344	1	6	5
500	497.9	497.5	706	702	2	5	5
500	500.2	500.2	605	608	0	10	10

control requirements for the second stage of a two-stage vehicle should be significantly less than those for the first stage.

Specific impulse was seen in figure 29(b) to have little effect on a vehicle's velocity losses, but it can have a large effect on payload ratio. Payload ratio is the ratio of usable mass (in-orbit mass less propulsion inerts) to launch mass. Figure 31 gives plots of payload ratio as a function of specific impulse for single-stage dual-burn vehicles and two-stage vehicles. The mass of propulsion inerts is assumed to be proportional to expended-propellant mass. An efficient design (low value of K) is seen to be especially important for the lower value of I_{sp}. The advantage of the two-stage vehicles is most clearly seen at the higher values of K and lower values of I_{sp}.

Because velocity losses are very sensitive to T/M (see fig. 29(a)), payload ratio data were generated for vehicles utilizing liquid propellant engines, whose inert masses are sensitive to T/M, and vehicles utilizing solid propellants. The following relationships which are typical were used to compute inert propulsion masses:

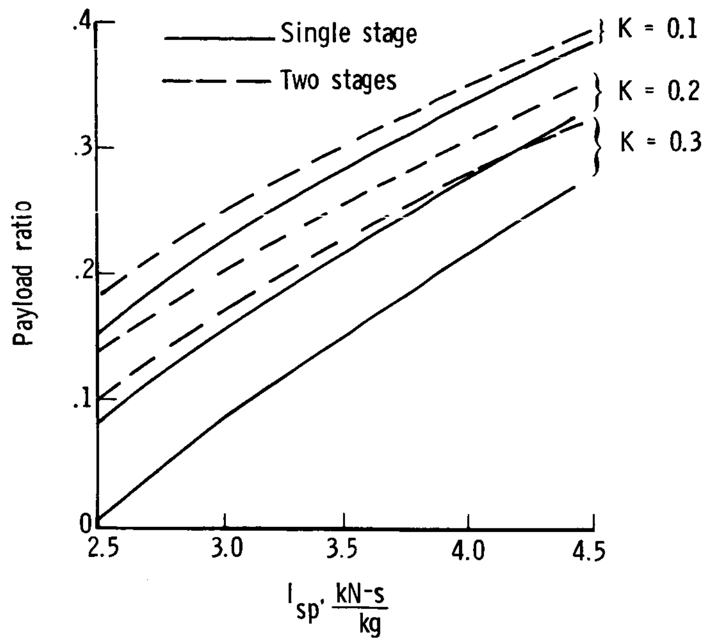
$$M_I = \begin{cases} 0.10M_p & \text{(Solid propellant)} \\ 0.145M_p + 0.00388T \dfrac{\sec^2}{m} & \text{(Liquid propellant)} \end{cases}$$

where

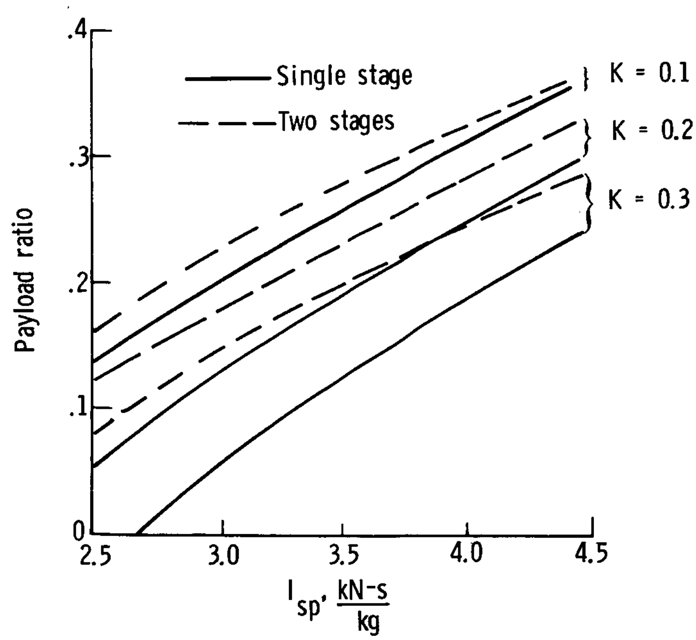
M_I mass of propulsion inerts

M_p mass of expended propellant

T thrust



(a) 100-km circular orbit.



(b) 300-km circular orbit.

Figure 31.- Effects of propulsion inerts and specific impulse on payload ratio for two-burn MAVs. Mean drag; V_1 = 2286 m/s; T/M = 5 Mars g units.

Data are presented in figure 32 for single-stage continuous-burn vehicles, and two-burn vehicles. No data are shown for single-stage dual-burn vehicles with solid propellants because restarting solid-propellant engines is not practical. The main data were generated by assuming a value of I_{sp} of 3.09 kN-s/kg for both types of propellants. Data are shown, however, for a solid-propellant vehicle with a specific impulse typical of a high-performance solid propellant. The effect of T/M on inert propulsion mass significantly reduces the payload ratios for the vehicles with liquid propellant motors, and it also shifts the point of maximum payload ratio to lower values of T/M than for the solids. Even when the differences in specific impulse between liquid and solid propellants are taken into account, the solid-propellant vehicle is seen to produce the greater payload ratio.

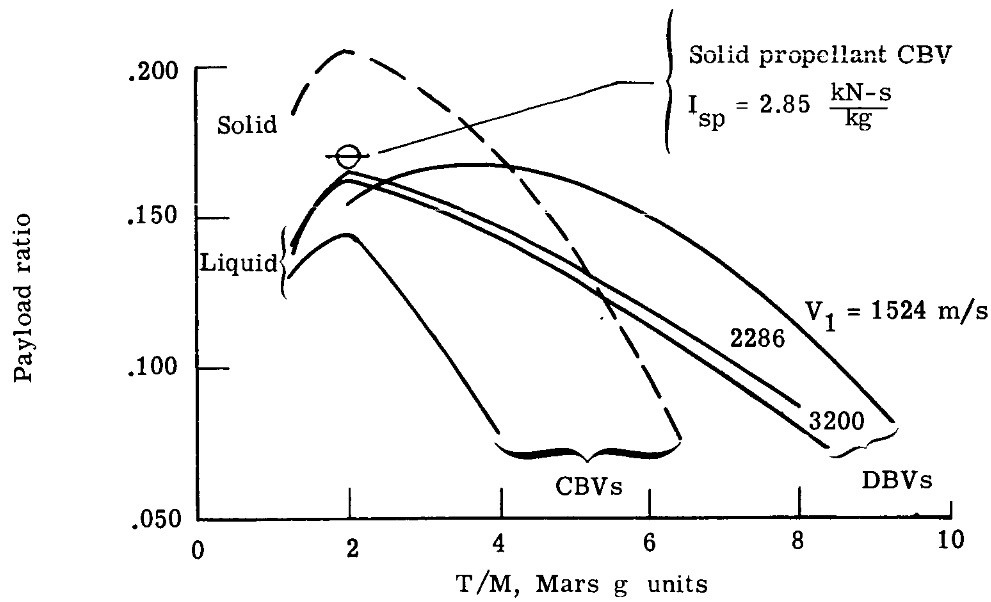
In summary, the study has identified the values of T/M and V_1 required for efficient single-stage dual-burn vehicles launched to Mars orbits, and these results were found to apply to two-stage vehicles. For the optimum trajectories, essentially all velocity losses take place during the first-burn phase. The second burn essentially thrusts horizontally to achieve orbit insertion with the result that the attitude-control requirements during the second-burn phase may be significantly less than those during the first-burn phase.

SYSTEMS DESCRIPTION AND OPERATION

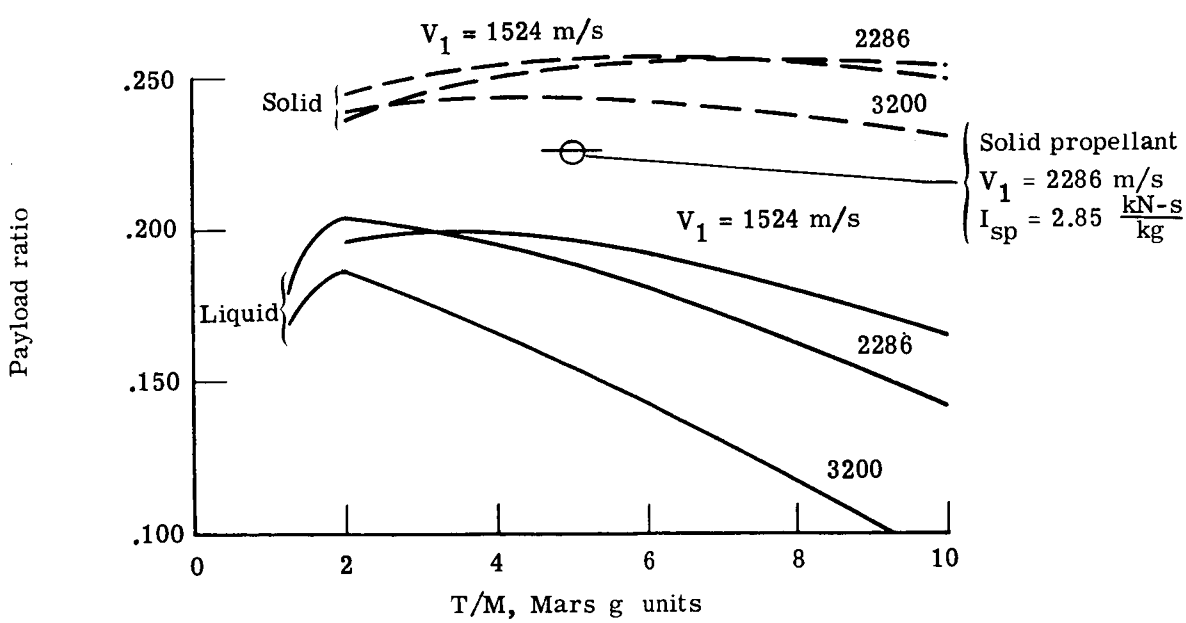
Baseline Mission

Because in the baseline missions all the systems required for the return trip to Earth are landed on and launched from the Mars surface, these missions place greater demand on configurational and packaging design of the MAV than does the MOR missions. Because of the greater mass which it must insert into orbit, the MAV for the baseline missions must also have more propulsion than the MAV for the MOR missions. On the other hand, the ERV, which has extensive telecommunication and data handling and command capability performs functions for the MAVs during the baseline missions which must be performed by the MAVs during the MOR missions. A sketch of the MAV for the baseline missions is shown in figure 33 and a functional block diagram of the MAV for the baseline missions is shown in figure 34. A summary of systems and masses for the baseline mission is given in table XIX. A scenario of the Mars surface-to-orbit operations for the baseline mission is shown in figure 35.

Propulsion.- The results of the study reported earlier in this section showed that MAVs employing solid propellant are superior to those using liquid propellant, and staging was also shown to increase the usable mass in orbit. Thus, two-stage solid-propellant vehicles were selected for the Mars surface-to-orbit phase for all missions. The requirements for configurational compatibility with the MEC and the desire to have a constant thrust during each entire thrusting period led to a baseline configuration with three motors



(a) Single-stage vehicles.



(b) Two-stage vehicles.

Figure 32.- Payload ratio data for single-stage and two-stage vehicles using solid and liquid propellant. $I_{sp} = 3.09 \frac{kN-s}{kg}$.

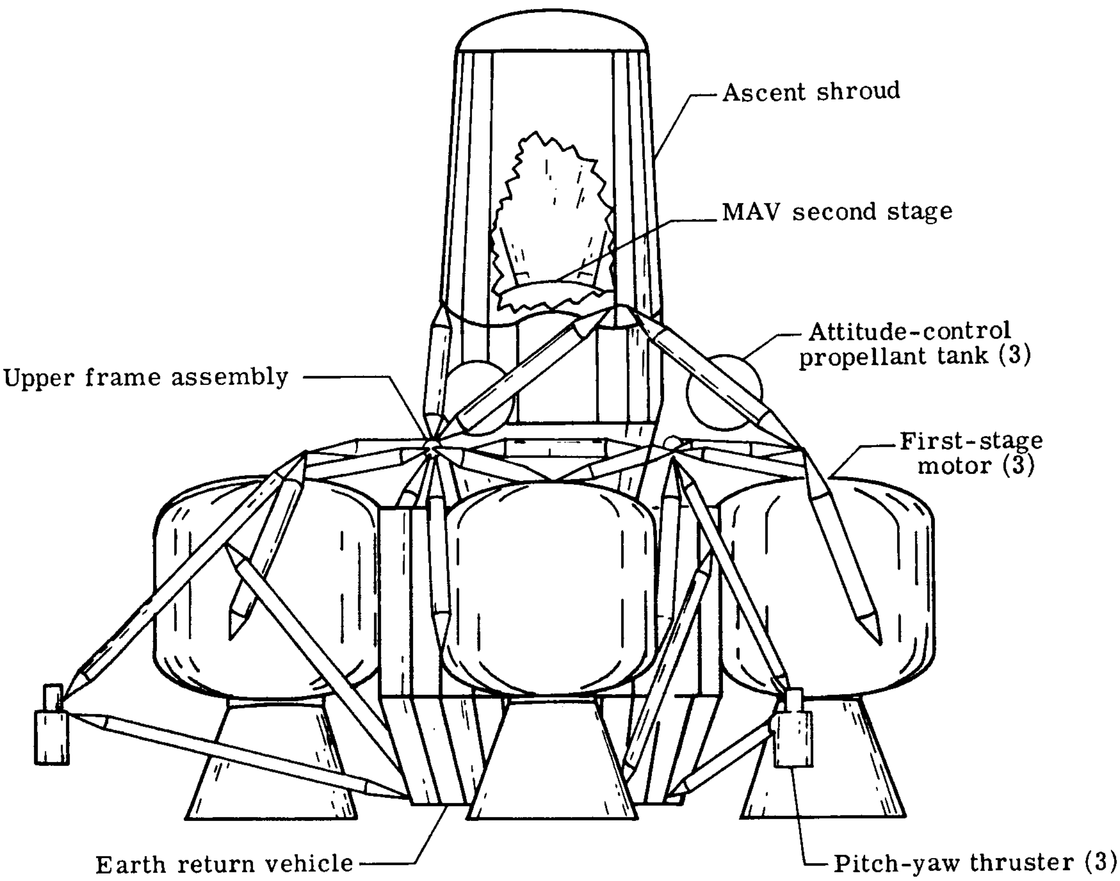


Figure 33.- Sketch of Mars ascent vehicle with Earth return vehicle for baseline mission.

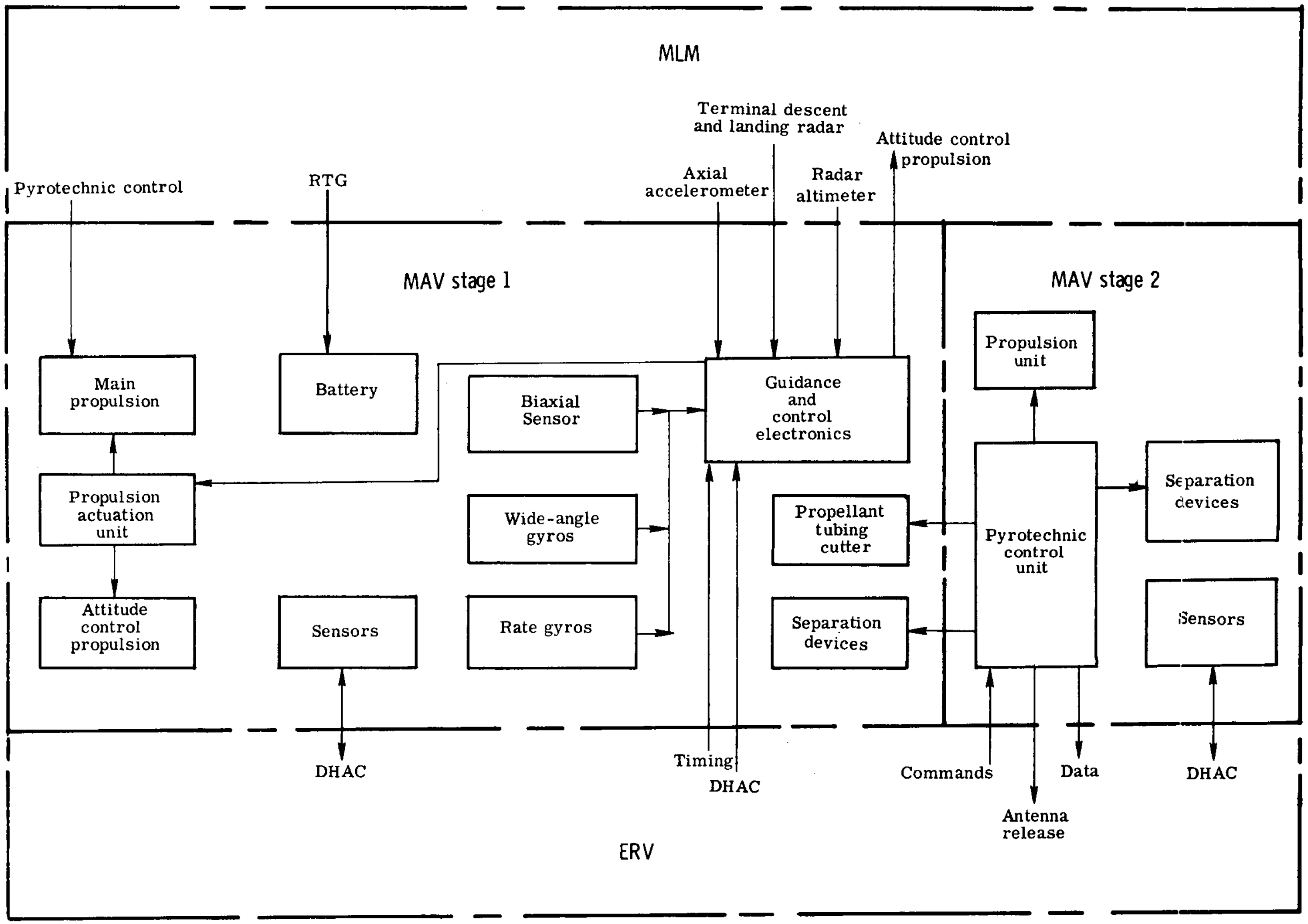


Figure 34.- Mars ascent vehicle functional block diagram illustrating MLM and ERV interfaces (baseline mission).

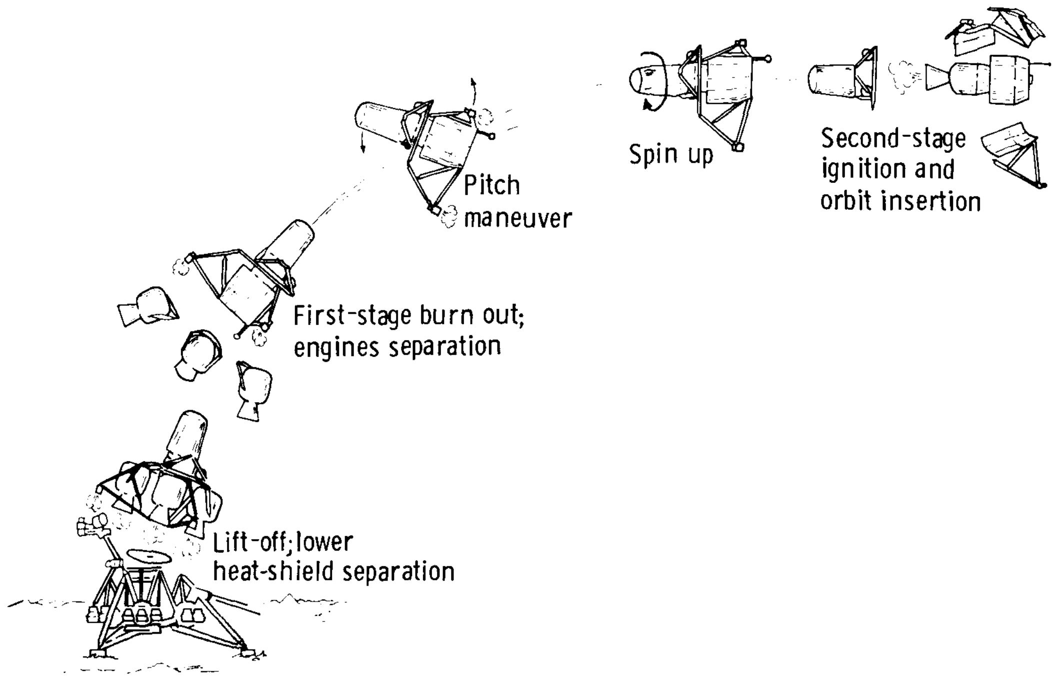


Figure 35.- Scenario of Mars ascent events. Baseline mission.

TABLE XIX.- SUMMARY OF SYSTEMS MASSES FOR MARS ASCENT VEHICLES

System	Mass, kg, for –					
	Stage 1			Stage 2		
	$(MAV)_1$	$(MAV)_2$	$(MAV)_3$	$(MAV)_1$	$(MAV)_2$	$(MAV)_3$
Structure	30.1	23.1	16.8	3.2	29.4	11.3
Telecommunications					8.5	8.6
Power	6.2		6.2		17.6	8.9
Data handling and command . . .					4.5	4.5
Pyrotechnics	8.7	5.0	4.5	5.4	6.4	5.4
Cabling	1.9	0.5	3.8	0.5	6.8	
Mechanical devices	5.9	4.5		0.4	6.7	
Temperature control	0.9	0.4	4.1		5.4	3.2
Attitude control (includes propulsion)	44.1		31.5		41.8	
Primary propulsion	990.1	674.0	354.8	280.4	230.4	87.8
Rendezvous and docking					5.0	4.1
Contingency					1.4	1.4
Total masses (stages)	1087.9	707.5	421.7	289.9	363.9	135.2
Total both stages	1377.8	1071.4	556.9			

on the first stage and one motor on the second stage. Packaging considerations also led to the necessity for the second stage of the MAV to be alined 180° from its normal direction for orbit injection. The reorientation of the MAV to its proper alinement is handled by the attitude-control system on the first stage of the MAV during the coast period.

Table XX lists the characteristics of the propulsion system final design for the baseline mission. The design of the motors, which took into account sterilization, and the specific impulse of the propellant were based on results of a rocket-motor design study reported in a later section. The velocity loss could be reduced by going to higher values of V_1 (see fig. 29), but this advantage would be offset by the additional mass required for the attitude-control system on stage 1. Figure 29 also shows that velocity losses begin to rise sharply between 200 and 300 km; to improve vehicle efficiency, the MAV inserts at the periapsis of a 225-km by 4765-km orbit. The attitude-control system of the ERV provides velocity to raise the periapsis to the required 300 km. The ERV maneuver is performed at apoapsis.

TABLE XX.- PROPULSION SYSTEM PARAMETERS FOR MARS ASCENT VEHICLE

(a) First stage

Parameter	Value for -		
	$(MAV)_1$	$(MAV)_2$	$(MAV)_3$
Thrust, N	29 025	19 884	10 320
T/M, Mars g units	5	5	5
Specific impulse, kN-s/kg	2.85	2.85	2.85
Ideal velocity, m/s	2369	2341	2413
Burn time, sec	86.5	85.9	87.4

(b) Second stage

Parameter	Value for -		
	$(MAV)_1$	$(MAV)_2$	$(MAV)_3$
Thrust, N	8807	6672	2335
T/M, Mars g units	5	5	5
Specific impulse, kN-s/kg	2.91	2.91	2.91
Ideal velocity, m/s	2139	2432	2249
Burn time, sec	82.5	88.7	83.9

Attitude control.- Results of the vehicle parameter and trajectory optimization study showed that essentially all velocity losses occur prior to ignition of the second-burn phase on an optimum trajectory from the Mars surface to orbit insertion. Since the ERV can correct orbit errors on the baseline mission that result from MAV errors, the MAV will have a three-axis control system on its first stage and will be spin-stabilized during second-stage thrusting. The attitude-control system on the MAV (stage one) will continue to operate during the coast period; just prior to periapsis it will properly aline the vehicle for second-stage thrusting and spin up the vehicle. The first stage will be jettisoned, and after the second stage burns out, it also will be jettisoned.

The MAV attitude-control system comprises a sensor package containing three rate gyros, three wide-angle gyros, and a two-axis pendulous sensor, the electronics package, the hot-gas thrust-vector control system (using hydrazine pressurized with helium) and its tankage, plumbing, valving, and three pitch/yaw thrusters as well as two roll thrusters. The thrusters are used not only for thrust-vector control during main motor thrusting but are also used to reorient the MAV/ERV combination for ignition of the MAV second stage. Additionally, the roll thrusters are used for spinup after the first-stage rocket-motor burnout and separation. In conjunction with additional equipment on the Mars lander module (MLM), the attitude control system on the MAV is also used for descent guidance and control.

Structure.- The MAV first-stage structure and thermal-control system consists of an upper frame assembly and support for the three motors, the five thrust-vector control thrusters, propellant and pressurant tankage, and the attitude-control electronic package. Also included are the ascent shroud, the three side panels and plume shields (for ERV protection) and a lower heat shield with separation fittings which is jettisoned 3 seconds after lift-off. The second-stage structure consists of the rocket motor support and the pyro control unit support.

Telecommunications, data handling and command.- The MAV contains no telecommunications or data handling and command (DHAC) equipment. The ERV telecommunications system takes care of all the telecommunication functions required by the MAV which include receiving the commands for all functions required during ascent. These commands are stored in the ERV DHAC system which issues them during ascent.

Power, pyrotechnics, and mechanical devices.- The sole power source for the MAV first stage is an 18-cell silver-zinc battery, having a capacity of 280 W-hr, which is kept charged by the MLM radioisotope thermoelectric generator until MAV lift-off. There is no power source on the MAV second stage; the only power user (pyro control unit) is powered from the ERV battery.

The first-stage pyrotechnics system comprises the propulsion actuation unit for the thrust-vector control thrusters, a propellant isolation valve and pressurant line-purge valve as well as propellant line-cutting guillotines (for the thrust-vector control system), all release devices, and the igniters and safe-arm circuits for the three solid-fuel rocket motors. The latter are fired by power from the MLM pyro control unit. All separation devices, the tubing guillotines, and pyrotechnic valves are fired from the pyro control unit located on the second stage; their firing commands come from the ERV DHAC system. Prior to first-stage separation, the isolation valve stops propellant flow from the tank, the line-purge valve admits helium (under pressure) to the lines and thrusters so that remaining propellant is expelled overboard, and the pyrotechnically operated guillotine cuts the propellant lines.

The second-stage pyrotechnics system consists of the pyro control unit, the rocket engine igniter with safe-arm circuitry, and the second-stage separation devices. In response to commands from the ERV DHAC system, the pyro control unit fires the main rocket engine igniters and the ERV antenna release device. The ERV antenna is deployed prior to MAV/ERV spinup.

The first-stage mechanical devices are a V-band for release of the attitude-control electronics battery assembly, upper frame release fittings and separation hardware, rocket motor release and separation hardware, and thrust vector control-assembly release and separation hardware.

MOR Missions

For MOR missions, the MAV inserts the sample canister into an orbit for rendezvous with the MOV and for sample transfer to the ERV. Vehicles with two types of attitude control are used. One employs spin stabilization on the second stage as does the baseline MAV. The other has a three-axis attitude-control system on both stages. The second stages for both vehicles have systems for communicating with the Earth and they have radar transponders and optical systems which assist the MOV in the rendezvous and docking maneuvers. The MAV with the three-axis controlled second stage is very similar to that for the baseline mission. A sketch of the MAV with the spin-stabilized second stage is shown inside the MEC in figure 22.

Vehicle employing three-axis attitude control on second stage.-

Propulsion: The MAV has three solid-propellant motors on the first stage and one on the second as is the case of the MAV for the baseline mission. Table XX lists the characteristics of the propulsion system final design. The required orbit is 872 km by 9800 km. (See table IV.) For efficiency the MAV first inserts at the periapsis of a 225-km by 9800-km orbit. The attitude-control system on the second stage provides the velocity (72 m/s) to raise the periapsis to 872 km.

Attitude control: The attitude-control system is located on the second stage and provides attitude control during the entire launch-to-orbit period as well as orbit-trim and rendezvous and docking maneuvers. The hydrazine monopropellant, pressurized with helium, feeds three pitch-yaw thrusters and two roll thrusters. The hot-gas hardware includes tankage and the required plumbing, valves, and transducers. Sun sensors provide Sun-referenced stabilization during ascent and the final rendezvous phase. Inertial guidance is provided by the same sensor package (gyros and pendulous sensor) described for the baseline mission MAV. This sensor package also assists with descent guidance and attitude determination prior to Mars lift-off.

Structure: The stage-one structure consists of the rocket motor support truss tubes and support fittings, the upper frame assembly, and the ascent shroud. Thermal control is provided by plume shields to protect the second stage during motor burn. A lower heatshield, with separation fittings, is jettisoned 3 seconds after lift-off.

The second-stage structure consists of the bus and electronics supports, supports for the attitude propulsion system thrusters and tankage, side panels for thermal protection during ascent, supports for the canister boom, antenna, rocket motor, and the solar panels.

Telecommunication, data handling and command: A telecommunication system essentially identical to the one on the ERV for the baseline mission is located on the second stage and provides all communication during Mars surface operations and during the rendezvous and docking maneuvers. It has two antennas identical to those on the Mars cruise module.

The DHAC system located on the second stage is identical to that used on the baseline mission ERV; it processes, stores, and distributes commands, and performs all data handling functions that the DHAC system on the ERV performs on the baseline mission. During Earth-to-Mars cruise, it stores commands received by the MOV DHAC system and reissues them during descent.

Power, pyrotechnics, temperature control, and mechanical devices: The power system is located on the second stage and comprises four solar panels, each having an area of 0.465 m^2 (5 ft^2); silver-zinc battery, battery charger, power switching and logic circuitry, power distribution, fuses, diodes, and the dc/ac/dc converter regulators providing regulated voltages for the radio; data handling and command; and the radio traveling-wave tube amplifiers. The total solar power furnished by the panels is 75 W at a Sun range of 1.6 AU. (The MAV equipment draws up to 73 W prior to the rendezvous maneuver.) The 18-cell battery has a 280 W-hr rating; its function is to take care of power transients and to provide power during the rendezvous maneuver.

The first-stage pyrotechnics system comprises the three rocket engine igniters with their respective safe-arm circuits, rocket motor separation devices, ascent shroud and upper ring separation devices, and lower heat shield separation devices. The first-stage mechanical devices are those used for rocket motor separation (two V-bands) and for upper frame release and separation (release fittings and separation hardware).

The second-stage pyrotechnics system consists of the pyro control unit, the propulsion actuation unit (to actuate attitude propulsion system thruster control valves), the separation devices for jettison of the first stage, and the igniter and safe-arm circuit for the main engine on the ERV. The second-stage temperature control is achieved by insulation shields, blankets, louvers, and heaters. Second-stage mechanical devices consist of the solar panel deployment mechanisms, solar panel dampers, the canister boom with its drive head and separation mechanism, and the canister insertion ram which is actuated after docking.

Rendezvous and docking: The system consists of the docking adapter (which mates with the MOV docking cone), a small radar transponder which operates in conjunction with the MOV approach-guidance radar, light sources with optics which operate in conjunction with the MOV X,Y optical trackers, a light sensor assembly which operates in conjunction with the MOV GaAs light sources to assist in MAV axis alinement for docking (backup mode), and the electronics required to interface the MAV rendezvous and docking system with the attitude propulsion and the data handling and control systems.

MAV employing spin-stabilized second stage.-

Propulsion: One solid-propellant motor is used on each stage of the MAV as primary propulsion. Table XX lists the characteristics of the motors for the final design. The MAV first inserts into a 225-km by 9800-km orbit. Two small solid-propellant motors located on the second stage raise the periapsis to the required 872 km.

Attitude control: The attitude-control system is scaled down from the one on the baseline MAV and performs the same functions which include orientation and spinup of the second stage prior to separation and ignition. The guidance and control package is identical to the one on the baseline mission and is also used during the Mars descent phase. The second stage will continue to be spin stabilized during the rendezvous phase and is despun with two small solid-propellant motors just prior to separation of the sample canister from the second stage.

Structure: The first-stage structure consists of equipment to support the thrust-vector control thrusters, propellant and pressurant tankage, the altitude control and sensor electronic package, and the rocket motor. A fairing and a lower heat shield are included. The second-stage structure includes supports for the electronics package, the antenna, and the rocket motors.

Telecommunications, data handling and command: The MAV first stage has no telecommunications system. A system on the MLM provides all necessary communications with Earth while on the Mars surface including receiving the ascent program. A communications system on the second stage provides communication with the Earth and the MOV during rendezvous; it is not activated until after orbit insertion. Because of the limited time during which the telecommunications system on the second stage will operate, its power source is reduced from that on the MAV with the three-axis controlled second stage. Command for initiating ignition of the motor on the second stage which raises periapsis is transmitted from Earth as is the command for despin and separation of the sample canister.

The second-stage DHAC system performs functions required during ascent and rendezvous, and it also provides the same functions during descent and surface operations that the DHAC system on the ERV provides for the baseline mission. As with the other missions a partial DHAC system located on the MLM functions in conjunction with the DHAC on the MAV second stage.

Power: An 18-cell 280 W-hr battery identical to the one on the MAV second stage of the baseline mission provides power from lift-off to separation. It is charged prior to launch by a charger on the MLM. Power for second stage includes a battery, distribution system, and a converter-regulator system.

Pyrotechnics and temperature control: The pyrotechnics system on the first stage includes the propulsion actuation unit, the propellant isolation valve, and components to ignite the rocket motor and to release the lower heat shield. The second-stage pyrotechnics system includes a pyro control unit, separation devices, and igniters for the rocket motors (main propulsion, periapsis raising motor, and despin motors). Temperature control on the second stage includes insulation, louvers, and heaters. A mechanical device on the first stage provides first- to second-stage separation, and a device on the second stage provides separation of the sample canister.

Rendezvous: The system is the same as the rendezvous and docking system for the MAV with a three-axis-controlled second stage except that no docking adapter is required.

SOLID-PROPELLANT ROCKET MOTOR DESIGN STUDIES

A heat-sterilizable solid-propellant motor design was studied for use on the MSSR missions. By appropriate scaling, the design could be used on the MAV, ERV, EEC, or EOC to reduce development and qualification costs. A representative motor design is shown in figure 36. The propellant grain is contained within a rubber boot which is secured in the case by flexible rubber seals. Thus, the grain is free floating within the case to accommodate the stringent heat sterilization and thermal environment requirements of the mission. A silicone fluid fills the volume between the propellant grain boot and case insulation and cushions the grain as it is positioned against the forward closure by compressed nitrogen. The pressure of the gas in the free volume of

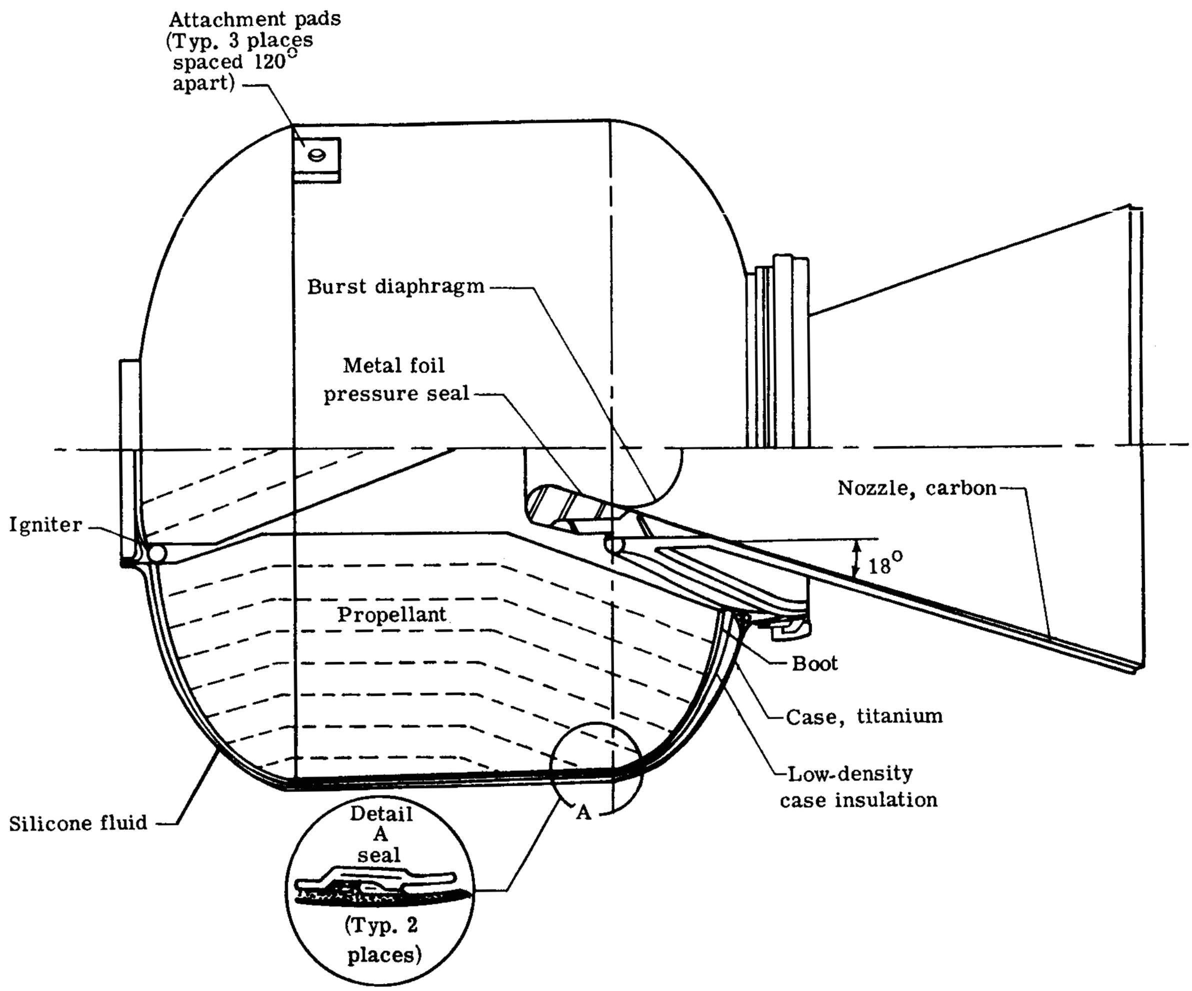
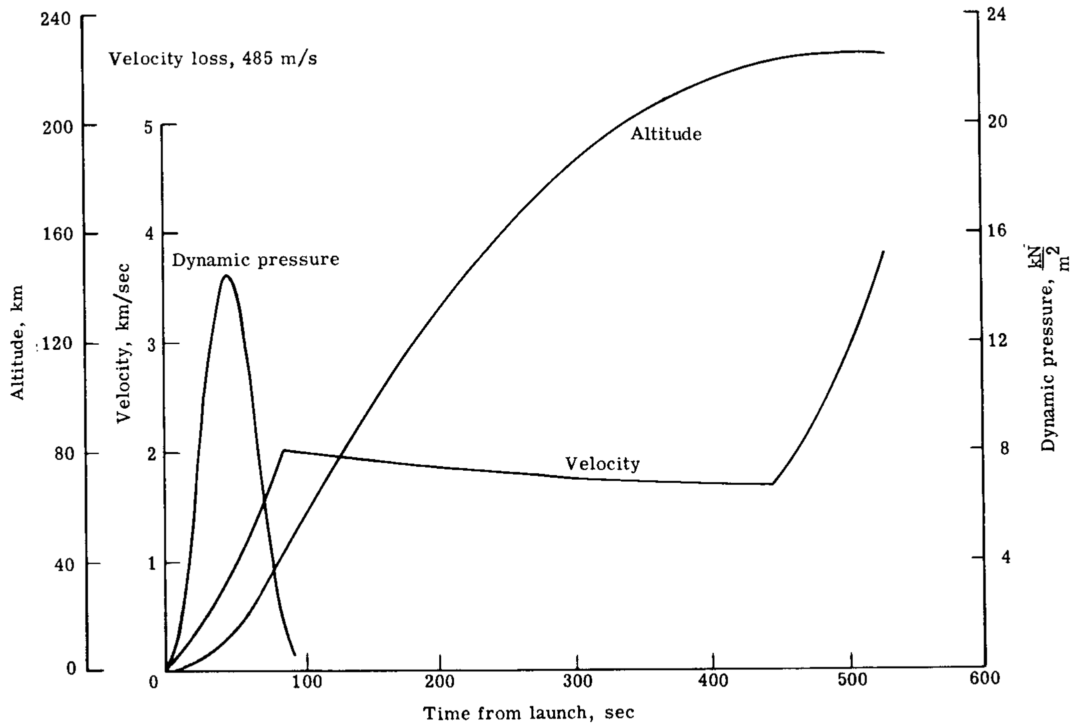


Figure 36.- Sketch of typical heat-sterilizable solid-propellant rocket motor.

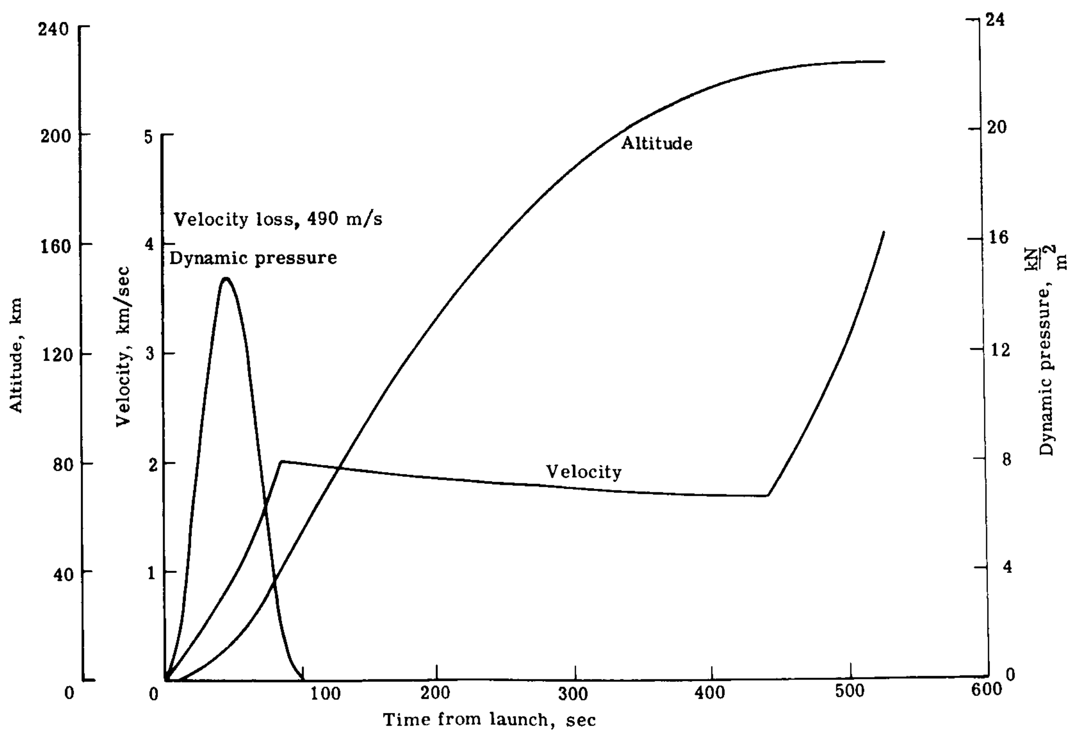
the motor is chosen to balance dynamic forces encountered by the propellant grain during normal mission life. The motor case is titanium and the submerged nozzle is a carbon composite. The nozzle closure and pressurant gas seal is a burst-diaphragm type.

ASCENT TRAJECTORIES AND VELOCITY LOSSES FOR THE FINAL-DESIGN MAVs

Trajectory data and velocity losses are presented in figure 37 for the three final-design MAVs launched on optimum trajectories to the required orbits. Launches were made eastward from a latitude of 35° into orbits of 35° inclination. Where appropriate, specific impulses were reduced to account for the propellant required for attitude control. The Viking mean density atmosphere was used. Values of initial M/C_DA are given

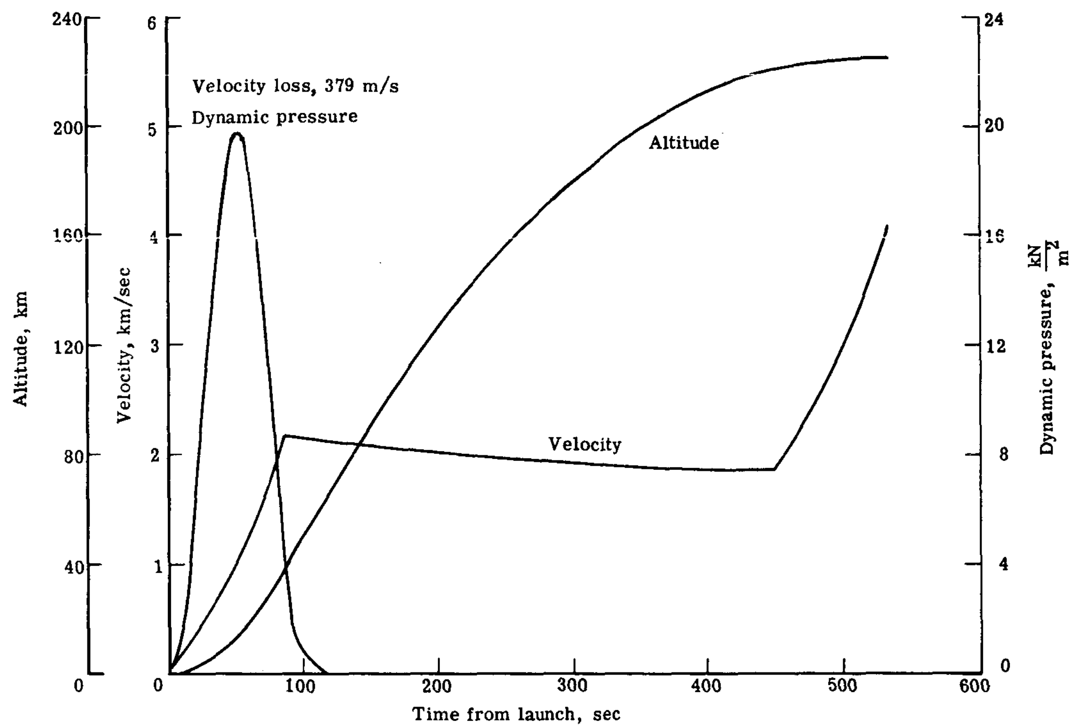


(a) Baseline mission.



(b) MOR; three-axis-controlled second stage.

Figure 37.- Trajectory parameters and velocity losses for final-design MAVs.



(c) MOR; spin-stabilized second stage.

Figure 37.- Concluded.

for the three vehicles along with the values of the parameters used in the mean drag (standard) cases during the trajectory and optimization study discussed earlier. The

	Baseline	MOR, three axis	MOR, spinner	Standard
M/C_DA, kg/m2	834	894	2413	2060

higher losses for the MAVs with three motors on their first stages are due to the higher drag on these vehicles.

It can be seen in figure 30 that the pitch program (which is typical) during the first-stage thrusting is nearly linear. Several minutes will be available during coasting for alinement and spinup (if appropriate) of the second stage.

SYSTEMS DESIGN AND OPERATION FOR PHASE-TWO ORBITING, MARS DEPARTURE, AND MARS-TO-EARTH CRUISE

The Earth return vehicle (ERV) is landed on and launched from the surface of Mars for the baseline missions, and it is inserted into Mars orbit with the Mars orbit vehicle (MOV) for the Mars orbit rendezvous (MOR) missions. It performs all operations during phase-two orbiting Mars departure and Mars-to-Earth cruise. Phase-two orbiting is that period after separation from the MOV on the MOR missions and is the entire orbiting time for the baseline mission. The major operations during these phases include maintaining attitude control, performing interplanetary trajectory and Mars orbit maneuvers, launching on to the interplanetary trajectory from Mars orbit, and providing communications with Earth.

The designs of most of the systems on the ERV are essentially identical for the baseline and MOR missions. The telecommunication, data handling and command (DHAC), and attitude-control electronics systems are also approximately identical in mass for the two missions, but on the baseline mission they play important roles during Mars descent, surface operations, and ascent as well as while in orbit. The requirements for the propulsion (including attitude control) system are less demanding on the MOR missions, however, and result in a total ERV mass less than that for the baseline ERV. Table XXI is a summary of systems and masses for the three ERVs for which systems were conceived. A sketch of the ERV is shown in figure 38, and a functional block diagram of the ERV for the baseline mission is shown in figure 39.

TABLE XXI.- SUMMARY OF SYSTEMS MASSES FOR EARTH RETURN VEHICLES

System	Mass, kg, for –		
	$(ERV)_1$	$(ERV)_2$	$(ERV)_3$
Structure	14.1	14.1	14.1
Telecommunications	7.2	7.2	7.2
Power	16.6	16.6	16.6
Data handling and command	4.5	4.5	4.5
Pyrotechnics	4.8	4.8	4.8
Temperature control	4.5	4.5	4.5
Mechanical devices	3.3	3.3	3.3
Cabling	3.2	3.2	3.2
Attitude control less ΔV propulsion	14.3	14.3	14.3
ΔV propulsion	17.3	8.1	8.7
Propulsion for Mars departure	78.5	65.7	67.9
Contingency	1.1	1.1	1.1
Back-contamination bioshield			9.1
Total	169.4	147.4	159.3

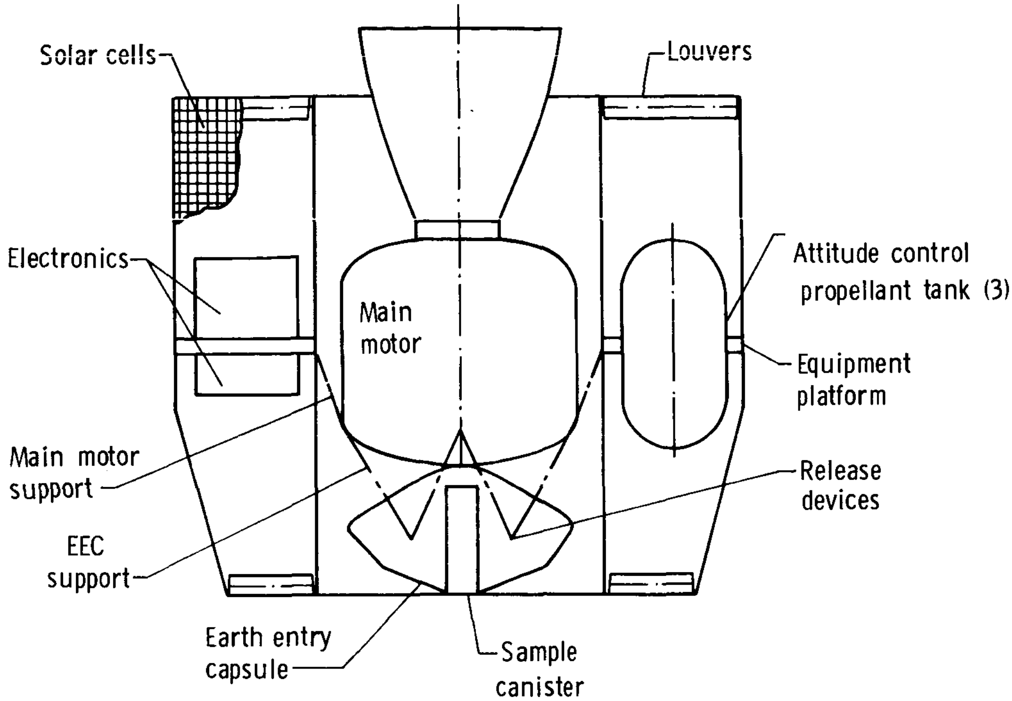


Figure 38.- Sketch of Earth return vehicle.

STRUCTURE

The ERV structure, as well as electronic packaging philosophy, is based on Pioneer 9 design. The electronics units are mounted to an annular equipment platform. The quasi-cylindrical external surface also provides substrates for solar cells. The thrust axis of the rocket motor for Mars departure is on the ERV center line. Supports are also provided for the Earth entry capsule (EEC) and the Earth orbit capsule (EOC).

PRIMARY PROPULSION

The primary propulsion system is comprised of a single solid-propellant rocket motor which provides the velocity increment for Mars departure. Its design features are the same as those for motors on the Mars ascent vehicles (MAV) which include heat sterilizability and long shelf life.

ATTITUDE CONTROL

The ERV attitude-control system maintains attitude control from the beginning of phase-two orbiting to Earth arrival. It also is used to raise the periapsis of the Mars orbit and to make orbit corrections on the baseline mission, and the guidance and control

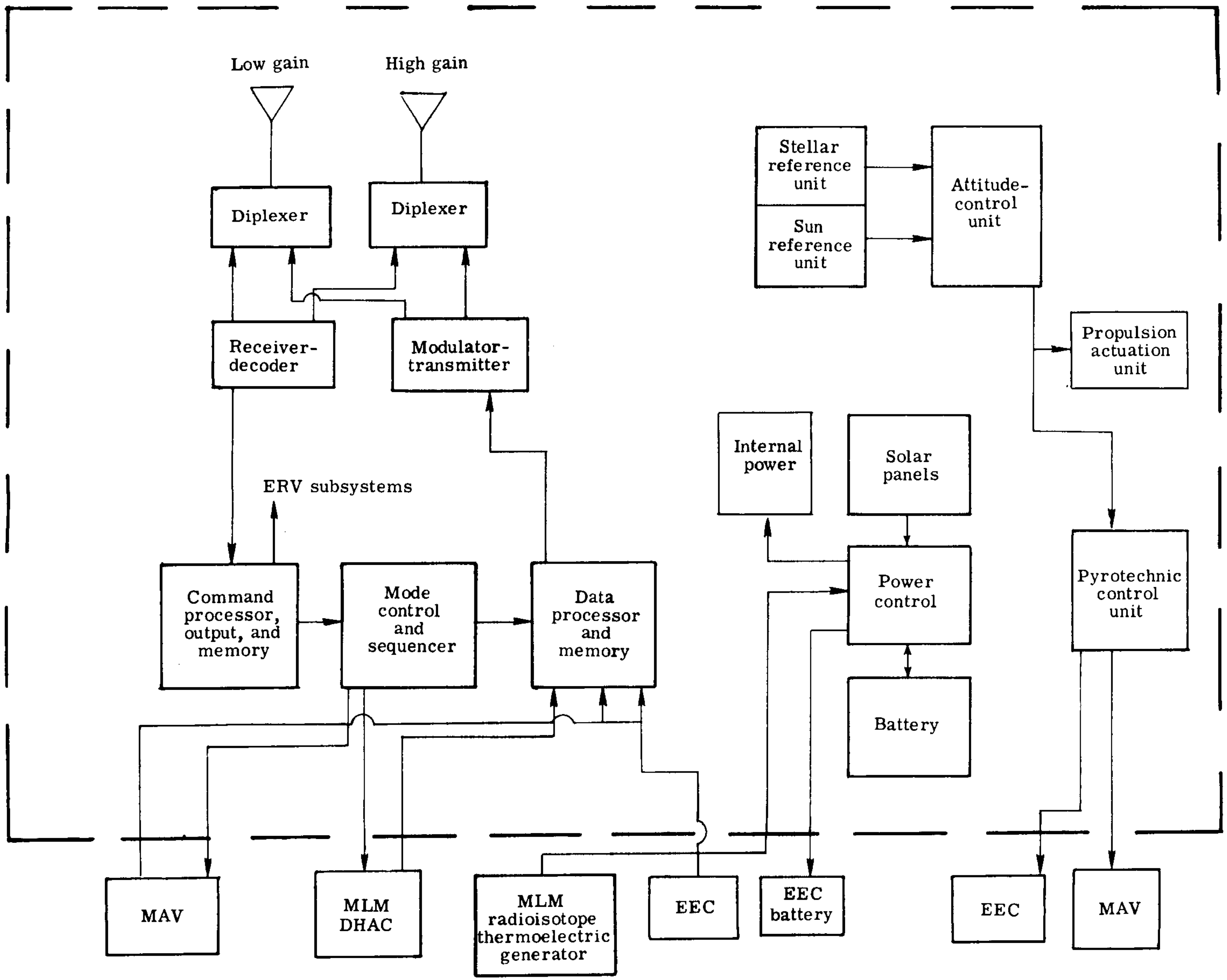


Figure 39.- Earth return vehicle functional block diagram for baseline mission.

package is used on the baseline mission during Mars descent and ascent. The system is based on that used in Pioneer 10 and 11 spacecraft. Spin stabilization is employed, the ERV spin axis being normal to the ecliptic plane. The nominal spin rate is 60 revolutions per minute which provides high gyroscopic stiffness and is consistent with structural and optical sensor limitations. As in the Pioneer 10 and 11 missions, all attitude-control functions are performed in an open-loop manner by ground command. The attitude references are the Sun and the star Sirius. The attitude-control system will maintain the spin axis to within $\pm 1.0^{\circ}$ of its commanded position so that the overall pointing requirement of the high-gain antenna ($\pm 2.0^{\circ}$) can be met. The six hot-gas thrusters are used for attitude control as well as for velocity-correction maneuvers; these thrusters use hydrazine in a blowdown system pressurized with helium.

The major electronics elements of the attitude-control system are the following:

(1) Control electronics assembly: It processes sensor input commands and data and produces outputs to fire appropriate precession thrusters and spin control thrusters. It controls power to all units within the control electronics assembly and provides processed sensor pulses and telemetry outputs.

(2) Sun sensor assembly: It provides Sun pulses to establish spin-axis direction about an axis normal to the Sun line. It also establishes a reference plane for open loop precession and provides a measure of spin rate.

(3) Stellar reference assembly: It provides star (nominally Sirius) pulses to establish spin-axis direction about an axis parallel to the Sun line. It also establishes a reference plane for open loop precession and provides a measure of spin rate.

The propellant supply and distribution elements supply hydrazine to the four precession thrusters and the two spin control thrusters through solenoid valves which are opened in response to commands from the control electronics assembly to provide spin, ΔV, or precession control. A propulsion actuation unit (part of the pyrotechnics system) is used to convert the command signals from the control electronics assembly into currents sufficient to energize the solenoids. The three propellant tanks are manifolded at both their pressurant and propellant ports. Two redundant latching valves are employed to isolate the propellant from the thrusters manifold. These propellant isolation valves are opened prior to Mars lift-off. The propellant for the baseline ERV includes that required to provide the velocities for Mars orbit corrections and interplanetary trajectory changes and to raise the Mars orbit periapsis (total $\Delta V = 257$ m/s). The velocity allocation for Mars orbit changes on the MOR missions was reduced from that on the baseline missions because the MOV will have made corrections prior to its separation.

The Sun sensor assembly and stellar reference assembly are located on the ERV so that attitude error information can be derived about two orthogonal axes normal to the spin axes. The Sun provides a reference for precession and a measure of spin rate. The star

Sirius provides a reference for error about the Sun direction and also provides a measure of spin rate. Both sensors use a slit detector arrangement. As the ERV rotates, the time between initial slit crossings (spin period) is proportional to spin rate whereas the time between crossings of the sensor slits is proportional to attitude error. Sirius was selected as the nominal star reference because it is the brightest star (X2 Canopus) and is favorably located in the celestial sphere. Other stars, however, can also be selected and are listed below.

Star	Celestial latitude	Brightness
Achernar	-60°	1/2 Canopus
Canopus	-75°	1
Rigel	-31°	1/2 Canopus
Vega	$+61^{\circ}$	1/2 Canopus

The ERV attitude-control system provides the following modes of operation:

(1) Fixed angle precession: A single precession pulse is initiated at one of four fixed angles: 0°, 90°, 180°, or 270°, following each command from the ground

(2) Real-time precession: Real-time firing is initiated by ground command and fires a preselected valve pair for a preselected duration each time a command is received

(3) Programed precession: A precession pulse is generated once per revolution at a programed angle for a programed period of time

(4) Programed ΔV: A continuous firing to two velocity precession thrusters for a programed length of time

(5) Real-time ΔV: Same as real-time precession

(6) ΔV/spin control thruster maneuver: A ΔV/spin control thruster maneuver is enabled by ground command. In this mode, the two spin control thrusters are alternately fired at revolution angles of 0° and 180°. Their thrust provides ΔV in a direction normal to the spin axis

(7) Spin and despin: Spin or despin is initiated by ground command. Firing spin control thruster 1 increases spin speed and firing spin control thruster 2 decreases spin speed.

TELECOMMUNICATIONS

The ERV telecommunications system provides two-way communications with Earth from the beginning of phase two orbiting to Earth arrival. In addition, the ERV for the baseline mission provides the same capability during Mars surface operations. It has the capability to telemeter data to Earth at 8 or 16 bits per second during all these periods, but the 16 bits per second capability is utilized only during Mars surface operations or as required by ground-operation personnel.

Command reception is provided continuously, but the transmitter is turned on by ground command and then turned off by an onboard timer. Thus, data handling and associated ground operations are reduced and ERV power is conserved.

The design of the telecommunications system is based on the low-mass low-power devices used in Pioneer 10 and 11 radio design. Receiving, transmitting, and command decoding elements are redundant. The diplexer-switch arrangement allows either antenna (low-gain or high-gain) to receive or send. During Mars descent and landed operations the diplexers are switched to the Mars lander module high-gain and low-gain antennas. Each traveling-wave tube amplifier has a nominal output power of 8 W for an input electrical power of 26 W. Both command decoder units receive the baseband signal from the active receiver unit, and either may be activated to process the data bits. Demodulated command data are presented to the command decoder units. The low-gain and deployable high-gain antennas are of the types used on Pioneer 9. Since these antennas are completely spin symmetric and therefore require no despinning apparatus, they effect a significant mass savings. The high-gain antenna is a colinear broadside array with a peak gain of 10.7 dB and full 3-dB beamwidth of 5°. For the specified ERV pointing error of $\pm 2^{\circ}$, the high-gain antenna produces a pointing loss of about 1.25 dB. The low-gain antenna is a multislot omnidirectional antenna with a peak gain of -2.5 dB and full 3-dB beamwidth of 110°. Both antennas are designed to operate with linear signal polarization. The deep space network has the capability to receive and transmit linearly polarized signals.

By using the subsystem described, the ERV telecommunications link characteristics can be summarized as follows:

(1) Telemetry range with high-gain antenna: 2.6 AU at 8 bits per second, 2.2 AU at 16 bits per second

(2) Command range with low-gain antenna: 3.4 AU (64 m deep space network station, 100 kW)

(3) Telemetry data coding: convolutional, rate 1/2, constraint length 7

(4) Assumptions for telemetry range: 64 m deep space network with 3-Hz tracking loop (may require programing of ground oscillators or appropriate selection of transmission times). Pointing-error high-gain antenna: $\pm 2^{\circ}$

DATA HANDLING AND COMMAND

The ERV DHAC system performs all the required functions from the beginning of phase-two orbiting to Earth arrival. In addition, for the baseline mission it handles most of the requirements during Mars entry, surface operations, and Mars surface-to-orbit ascent and also plays a smaller role during Earth-to-Mars cruise. The overall requirements for the baseline mission are:

(1) Provide and maintain a central timing source for the ERV, MAV, and MLM

(2) Generate and issue internal commands to appropriate ERV, MAV, and MLM systems

(3) Receive and decode ground commands transmitted to the spacecraft for ERV, MAV, and MLM action

(4) Provide data format and sequencing control for the MLM DHAC system

(5) Receive, multiplex, digitize, sequence, buffer, and store incoming engineering data from the EEC, ERV, and MAV to provide a single data stream to the ERV modulation-demodulation subsystem

(6) Insert coding information in the data stream to the ERV modulation-demodulation system to identify each measurement and give the status of the DHAC system

Present DHAC designs exceed these requirements because this mission has no science instruments which require high data rates. Design concepts for a new system were derived for this mission which will be smaller, less massive, and require less power. DHAC systems identical to the one on the ERV are employed on the Mars cruise module for the baseline mission and the first stage of the MAV of the MOR missions.

POWER

Photovoltaic solar cells are used as power source. The requirements for the solar-cell array are dictated by power requirements and by the Sun distance shown in figure 4.

The quasicylindrical external structure of the ERV was chosen to be a dodecahedron providing substrates for 12 long, but narrow, solar panels which are connected in parallel, protected by a diode for each panel. A nickel-cadmium rechargeable battery is charged from a simple battery charger. The primary bus, in itself not closely regulated, feeds redundant dc/ac/dc converters which provide the regulated voltages needed by the various systems. When data transmission is commanded, one of two available traveling-wave tube amplifier converters is switched on (the converters are also selected by ground command) so that it can provide the high-voltage power required. Unregulated ("raw") direct current from the primary bus is furnished to such users as heaters and pyro control unit, as well as the MAV pyrotechnics system and to the battery in the EEC (or EOC) for trickle charging it. ERV power can also be obtained from or augmented by external power (during prelaunch operations) or by RTG power (prior to Mars lift-off). The system which distributes regulated voltages to system users is similar to the power distribution system used on Pioneer 9.

The ERV relies heavily on battery power during Mars ascent (when the solar array is not yet exposed to the Sun) and during telemetry transmissions when solar power alone would not be sufficient. The solar cell array area was sized at 2.8 m^2 (30 ft^2) and the battery capacity was selected as 3 A-hr.

THERMAL CONTROL, PYROTECHNICS, CABLING, AND MECHANICAL DEVICES

Temperature control elements include louvers, insulation, and main rocket motor nozzle cowling. Included are small heaters for the Sun sensor, the telecommunications transmitter oscillator (replacement heater), and the propellant lines.

The ERV pyrotechnics system consists of a pyro control unit and a propulsion actuation unit as well as two actuation devices. The pyro control unit ignites the motors for Mars departure and initiates separation of the EEC (or EOC). The propulsion actuation unit actuates the thruster control solenoid valves in response to pulses received from the control electronics assembly of the attitude-control system.

It should be noted that for the baseline mission the propellant isolation (latching) valves are actuated from the MLM pyro control unit and that the ERV antenna deployment is initiated from the MAV second-stage pyro control unit.

Two types of cabling techniques are used on the ERV. The intramodule cabling, which interconnects the various ERV systems, is hardwired without use of connectors. The intermodule cabling, which carries the electrical interfaces between the ERV and other modules of the MSSR spacecraft system, consists of wiring harnesses with connectors.

Mechanical devices onboard the ERV are used to deploy the antenna, to separate the second stage of the MAV on the baseline mission, to separate the EEC (or EOC), and to provide nutation damping.

SYSTEM DESIGN TO PREVENT EARTH CONTAMINATION

For two MOR missions, one of the design approaches discussed in reference 1 was implemented. On these missions, prevention of Earth contamination by Mars organisms is achieved by (1) surrounding the ERV external surface and all the EEC (or EOC) external surfaces up to the edge of the canister with a bioshield, and (2) sealing and heat-sterilizing the exposed surface of the canister, after insertion, with a combustible plug. This design approach results in a relatively small increase in the ERV mass over that of the ERVs for the other MOR missions (see tables XXI and XXXI) and the ERV mass for this mission is still less than that for the baseline mission.

EARTH ENTRY AND EARTH ORBITAL SYSTEMS DESIGN

Both direct entry and orbit capture modes were considered for the Earth arrival phase of all missions investigated. (See tables II and VI.) An Earth entry capsule (EEC) and its systems were designed and integrated with the Earth return vehicle (ERV) which is capable of entering a sample canister containing 200g of Martian material. An Earth orbiting capsule (EOC) and supporting systems were also designed which have about the same mass as the EEC and which were compatible with the ERV. This matchup in capsule masses resulted in the requirement for a highly elliptical Earth orbit but leads to the favorable result that all missions are essentially identical (including systems masses) from Mars departure to Earth arrival.

EARTH ENTRY MODE

At Earth arrival the ERV is on a trajectory which precludes Earth impact. Systems on the ERV provide initial separation of the EEC, and the EEC must then provide a deflection velocity that puts it on the desired entry trajectory. The EEC systems were designed to accomplish the scenario illustrated in figure 40. The deflection rocket motor uses solid propellant and is based on the rocket-motor study discussed earlier. It has a specific impulse of 2.91 kN-s and provides a deflection velocity of 100 m/s. It is ignited by a slow-burning fuse which is initiated by systems on the ERV prior to separation of the EEC from the ERV. After separation and deflection, the EEC will be decelerated by the atmosphere to subsonic velocity by the time it reaches an altitude of 20 km where a parachute is deployed to decelerate the capsule to a terminal velocity of approximately 40 m/s. The parachute design is based on current subsonic deployment technology. Parachute deployment will be based on static pressure. The capsule can be recovered by an air snatch at an altitude of 1.5 km as shown in figure 40, or recovered after landing in one of the oceans. A beacon is incorporated which can assist in aerial recovery or a water recovery.

The 1981 mission entry velocity of 12.5 km/sec was used for design. (See table III.) This velocity is above the Apollo reentry value for lunar return missions, and radiative heating will be significantly increased. To reduce radiative heating, a 120° cone with a

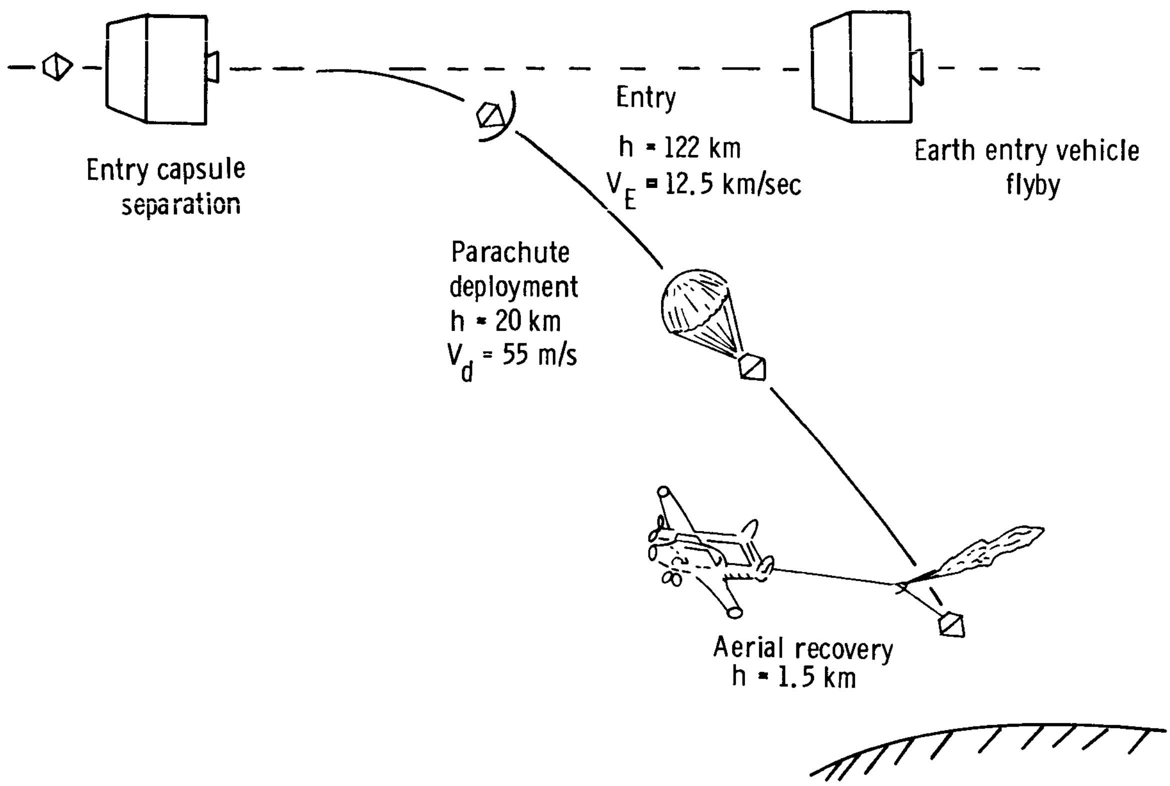


Figure 40.- Earth entry events.

small radius spherical nose was selected for the entry shape. Values of the maximum stagnation-point heating rates and the stagnation-point integrated heating are presented in figure 41 as a function of entry angle for a 120° total-angle cone having a nose radius of 7.55 cm and a base radius of 21.85 cm. The convective heating rates were defined by the cold-wall convective equation given in reference 17 and the radiative heating rates were calculated for an inviscid flow field with coupled radiation in accordance with equations developed in reference 18. The entry trajectories for the heating calculations were determined by using a ballistic coefficient M_E/C_DA of 74.1 kg/m^2, a zero lift-drag ratio, and an entry velocity of 12.5 km/sec.

As can be seen in figure 41, the convective heating obtained is significantly greater than the radiative heating. It is of interest to note that as the entry angle increases, the heating rates increase but the heating loads actually decrease. As a compromise, a nominal entry angle of -14° was selected, and the heat shield and structural design of the entry capsule are based on this angle. The structure was designed for a deceleration load of 85 Earth g units, the maximum value which is obtained on the -14° entry trajectory. The impact range dispersion produced by the entry corridor of $\pm 0.17^{\circ}$ is approximately ± 50 km, a reasonable spread for recovery purposes.

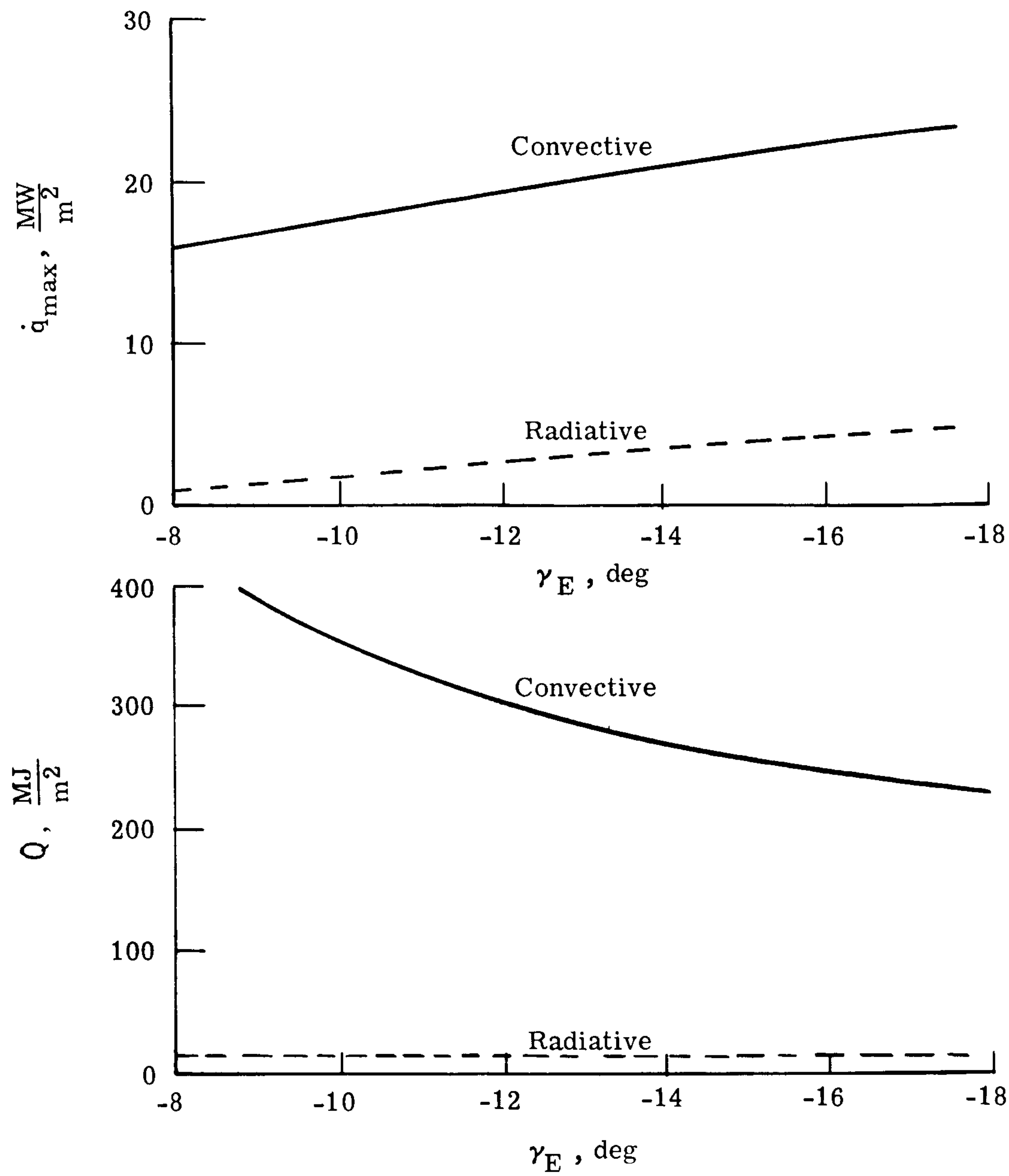


Figure 41.- Entry heating characteristics of the Earth entry capsule. V_E = 12.5 km/sec.

A summary of trajectory and vehicle parameters used in the design of the entry systems is given in table XXII, and an altitude-velocity profile for a typical entry is shown in figure 42. A sketch of the EEC which illustrates the location of its systems is shown in figure 43. Systems and masses are listed in table XXIII, and a functional block diagram is shown in figure 44. The primary systems are the heat shield, the structure, a beacon transponder, power, a sample canister, a 200-g sample, a parachute, and the deflection motor.

TABLE XXII.- DESIGN CONDITIONS FOR EARTH ENTRY CAPSULE

Parameter	Value
V_E, km/sec	12.5
M_E/C_DA, kg/m^2	74.1
L/D	0
γ_E, deg	-14 ± 0.17
$\dot{q}_{C,max}$, MW/m^2	22
$\dot{q}_{R,max}$, MW/m^2	4
$Q_{C,max}$, MJ/m^2	265
$Q_{R,max}$, MJ/m^2	15
Maximum Earth g	85

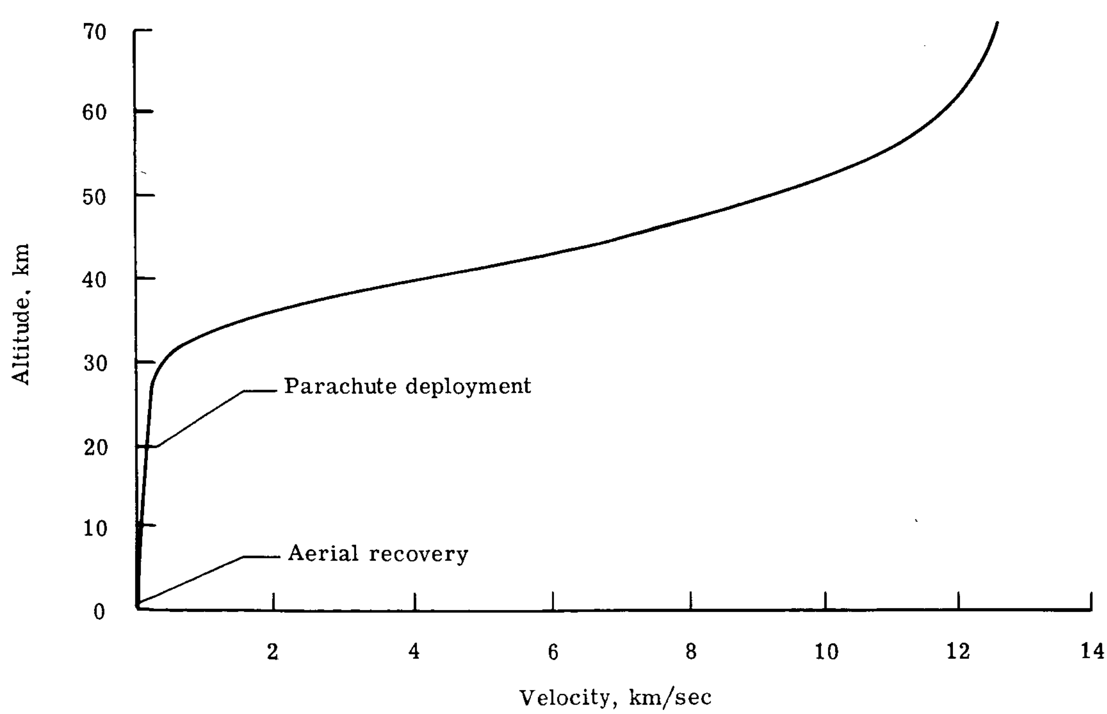


Figure 42.- Altitude-velocity profile for typical Earth entry. V_E = 12.5 km/sec; L/D = 0; M_E/C_DA = 74.1 kg/m^2.

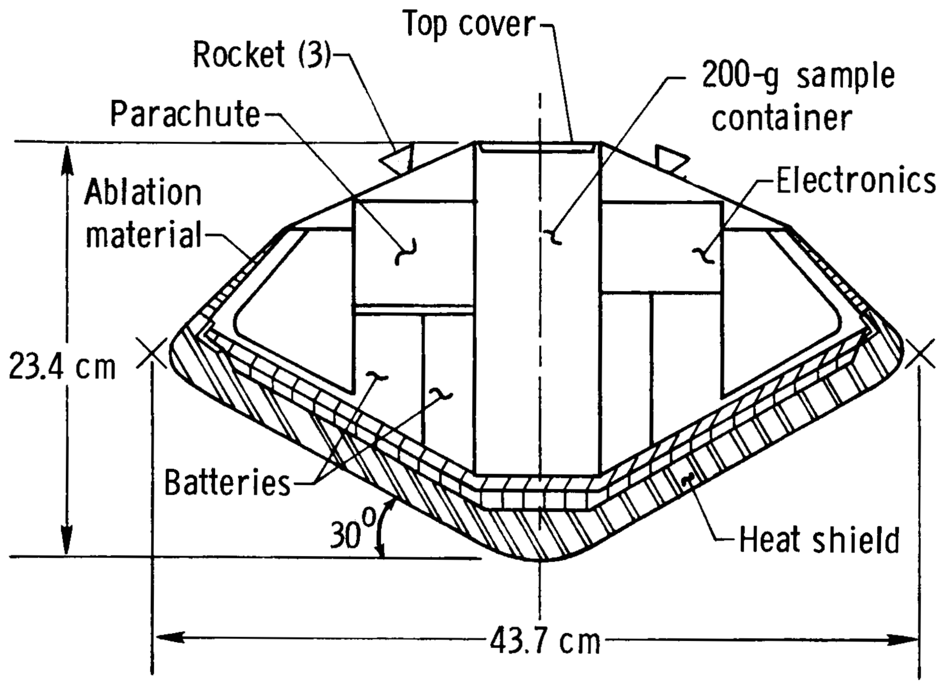


Figure 43.- Sketch of Earth entry capsule and its systems.

TABLE XXIII.- SYSTEMS AND MASSES FOR EARTH ENTRY CAPSULE

System	Mass, kg
Structure	4.2
Heat shield	4.0
Parachute	0.7
Telecommunication	1.8
Power	1.5
Sample canister	0.7
Sample	0.2
Propulsion	2.2
Total	15.3

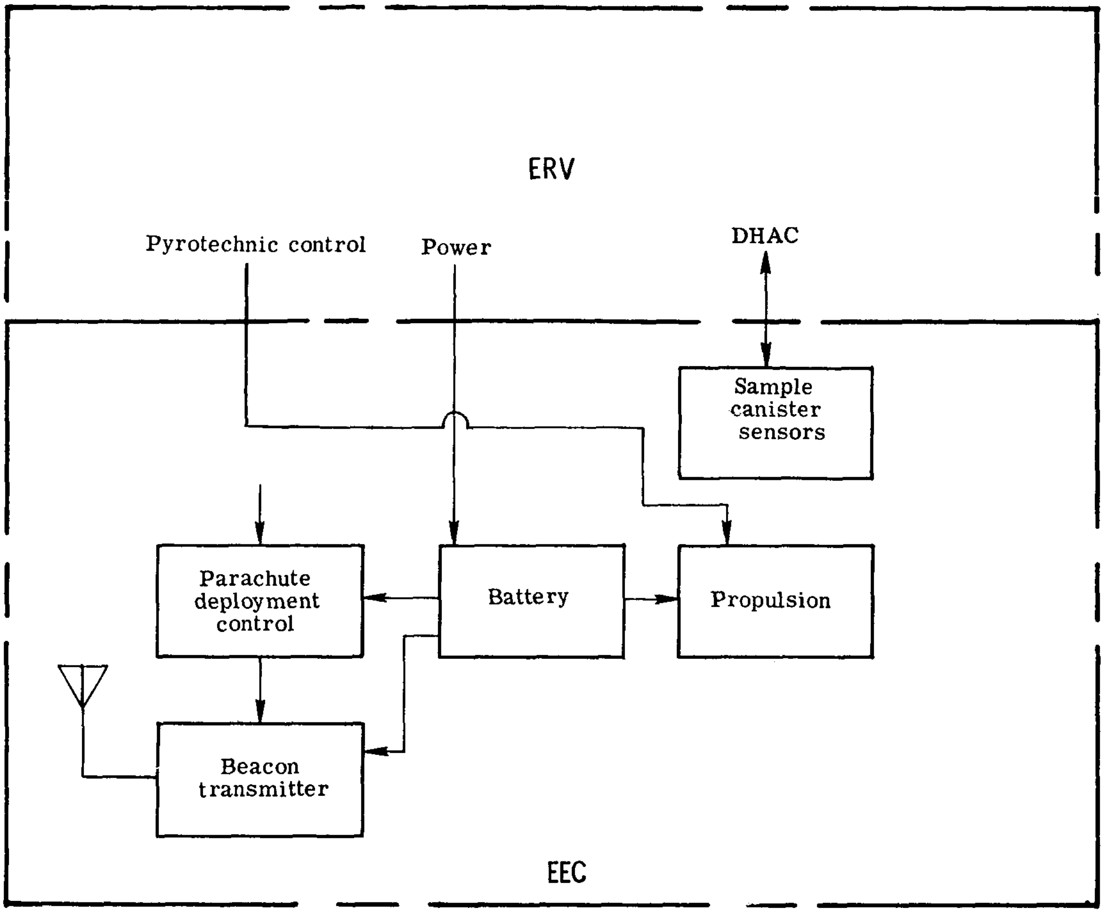


Figure 44.- Earth entry capsule function block diagram showing interface with ERV for the baseline missions.

ORBIT CAPTURE MODE

The trajectory conditions at Earth arrival are the same as for the Earth entry mode. Arrival velocity from table III is seen to be 4.6 km/sec. The ERV provides initial separation, but the EOC must brake itself into orbit. Subsequently, a vehicle such as a space tug could retrieve the sample and deliver it to the Earth or to an orbiting laboratory. A beacon transponder is incorporated to aid in locating the EOC in orbit and to assist in rendezvous and docking.

It was desirable to find an orbit so that the incorporation of a propulsion system would result in an EOC design with approximately the same mass as the EEC. Figure 45 shows the ΔV for establishing planetary return orbits at Earth as a function of eccentricity and apoapsis altitude for the 1979 mission. The periapsis altitude was assumed to be 500 km, which corresponds to the nominal operation orbit for the Space Shuttle. As can be seen, the velocity required for placing the return payload in an Earth orbit can be

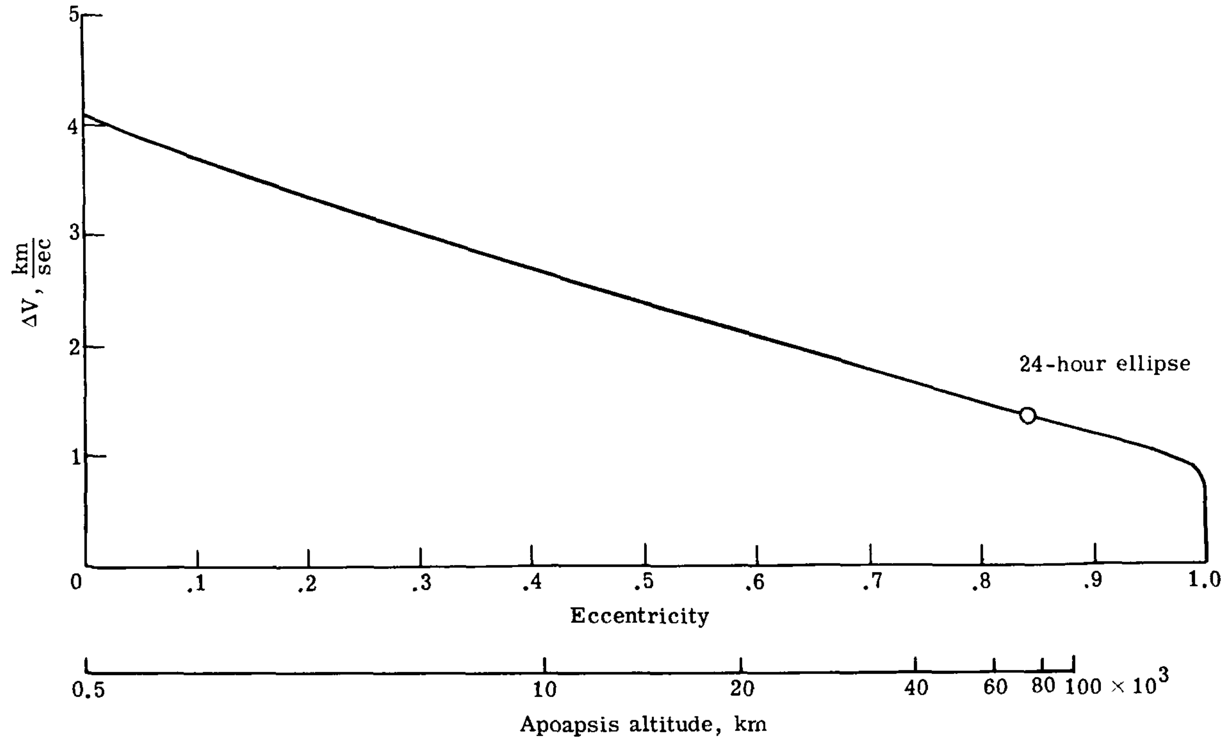


Figure 45.- Effect of orbit eccentricity on Earth capture ΔV. 1979 Earth launch; Earth arrival velocity = 4.6 km/sec; and h_p = 500 km.

reduced considerably by use of elliptical orbits. The 24-hour orbit was selected initially, and when the propulsion system necessary to provide the required braking velocity of 1.35 km/sec was defined, the resulting total mass of the EOC was found to be 16.1 kg which is very close to the mass of the EEC. The solid-propellant orbit insertion motors are the same design as the one on the EEC, and they are ignited in the same manner. The mass breakdown of the systems for the EOC is given in table XXIV, and a sketch of the EOC illustrating the locations of the systems is shown in figure 46.

TABLE XXIV.- SYSTEMS AND MASSES FOR EARTH ORBIT CAPSULE

System	Mass, kg
Propulsion	8.1
Structure	3.4
Telecommunication	1.8
Power	1.9
Sample canister	0.7
Sample	0.2
Total	16.1

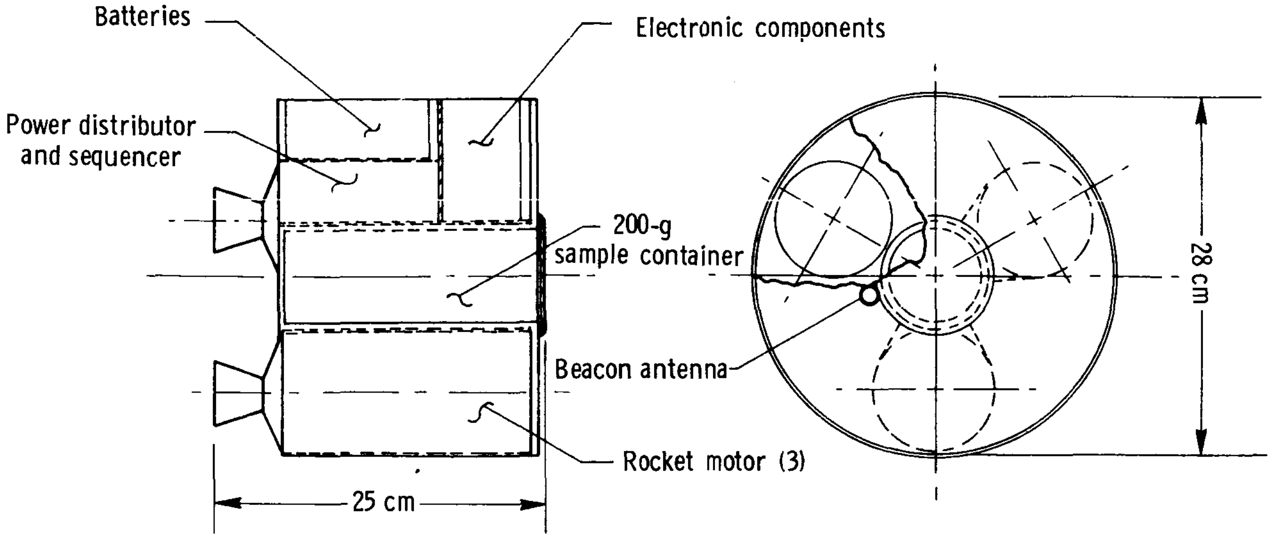


Figure 46.- Sketch of EOC.

Since the Space Shuttle will be developed by the time of the Earth return phase of the mission in 1982, it can be used in conjunction with a space tug to retrieve the sample payload in the elliptical orbit and return the sample to a Space Laboratory in a circular orbit. An interim tug such as a Centaur has at least the propulsive capability to transfer from a 500-km space shuttle orbit and rendezvous with a sample in a 24-hour elliptical orbit and then return it to the Shuttle/Space laboratory orbit. Reusable tugs which are being considered for the 1980's will also have the capability to retrieve the sample. Further study is required to define the complete capability.

MASS SUMMARY AND EARTH LAUNCH VEHICLE REQUIREMENTS

Summaries of system masses for major components are given in previous tables. These tables are indicated below for each component.

Component	Mass summary table number
MCM	VII
MOV	VIII
Aeroshell	XIV
Parachute systems	XV
Mars lander modules	XVI
Mars ascent vehicles	XIX
Earth return vehicles	XXI
Earth entry capsule	XXIII
Earth orbit capsule	XXIV

Total Earth launch masses and the contributions of components to the totals are shown in figure 47. The launch capabilities of the Titan III E/Centaur and Space Shuttle/Centaur are also shown in figure 47.

Ten of the twelve missions investigated can be performed with the use of a single Titan III E/Centaur, and all missions are possible with the use of the Space Shuttle/Centaur. The MOR missions which employ MAVs with spin-stabilized second stages have the lowest launch mass (not the baseline mission which is the minimum energy mission). The MOR missions which employ MAVs with three-axis-controlled second stages in conjunction with out-of-orbit Mars entry require the most launch mass.

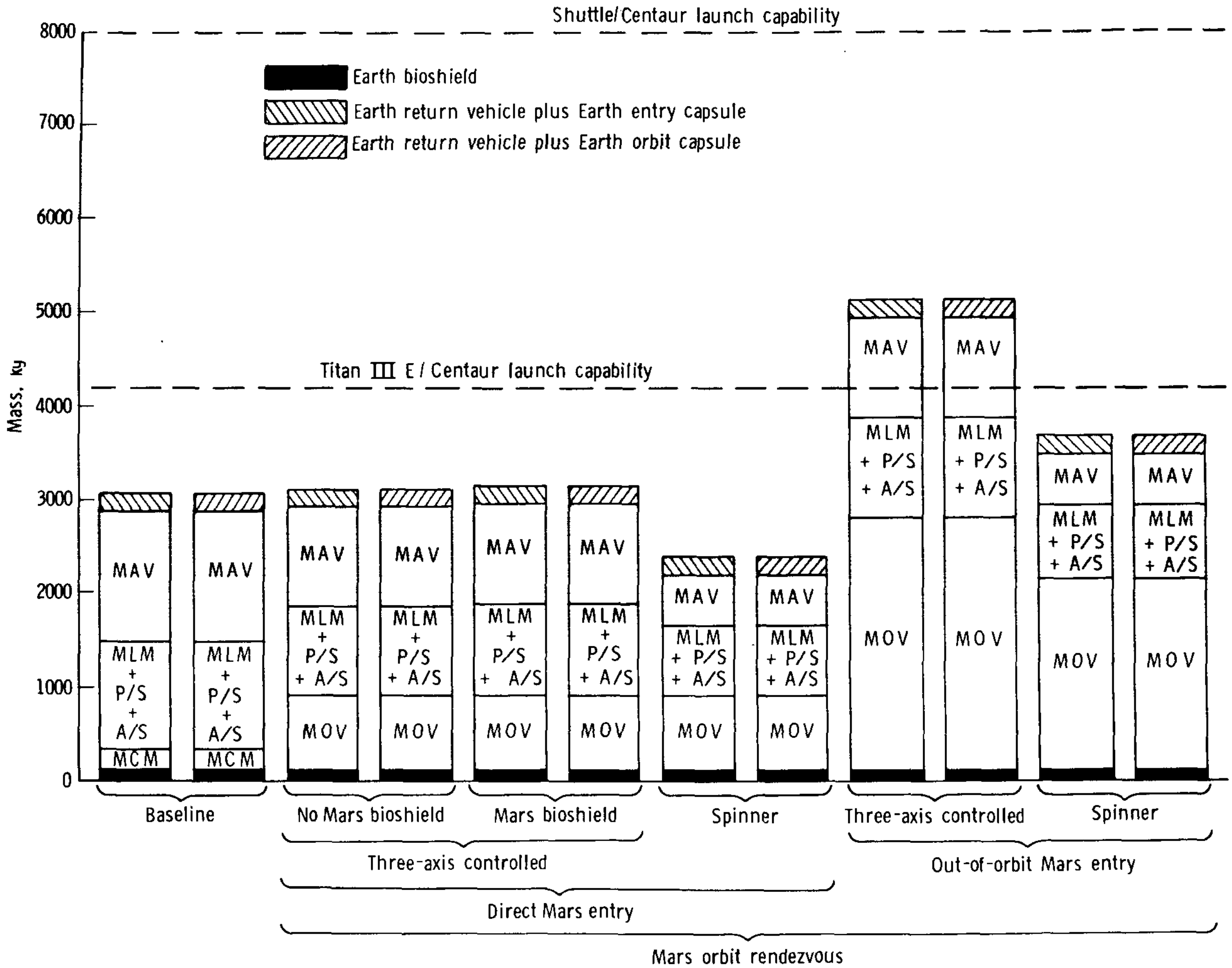


Figure 47.- Component and total launch masses for missions investigated.

APPLICABILITY OF RESULTS TO OTHER MISSION OPPORTUNITIES

Comparisons of the velocity and energy requirements for conjunction-class mission opportunities between 1975 and 1988 are presented in figure 48. As the launch year moves from 1979 and 1981 to 1984 and 1986, the geocentric injection energy increases from about 12 to 17 km^2/sec^2. With the use of the space shuttle and space tug in the mid-1980's, the higher values of C_3 can be easily accommodated. The entry velocity at Mars increases only from 5.8 to 6.1 km/sec as the launch year advances into the mid-1980's. This small increase has little effect on the entry mass as illustrated in figure 17 and produces little decrease in usable landed mass. The Mars departure velocity for 1984 is about the same as for 1979; however, it does increase for 1986 and 1988. Earth entry velocity is less in the mid-1980's than for 1979 and 1981. In summary, it appears that a 1984 mission could be readily accommodated with only minor changes to the 1979 and 1981 results. With greater modifications, all missions could be accomplished in the 1980s.

CONCLUDING REMARKS

Efficient mission concept approaches were investigated with the objective of acquiring and returning to Earth a Mars sample adequate for investigation in Earth laboratories.

SCIENCE REQUIREMENTS

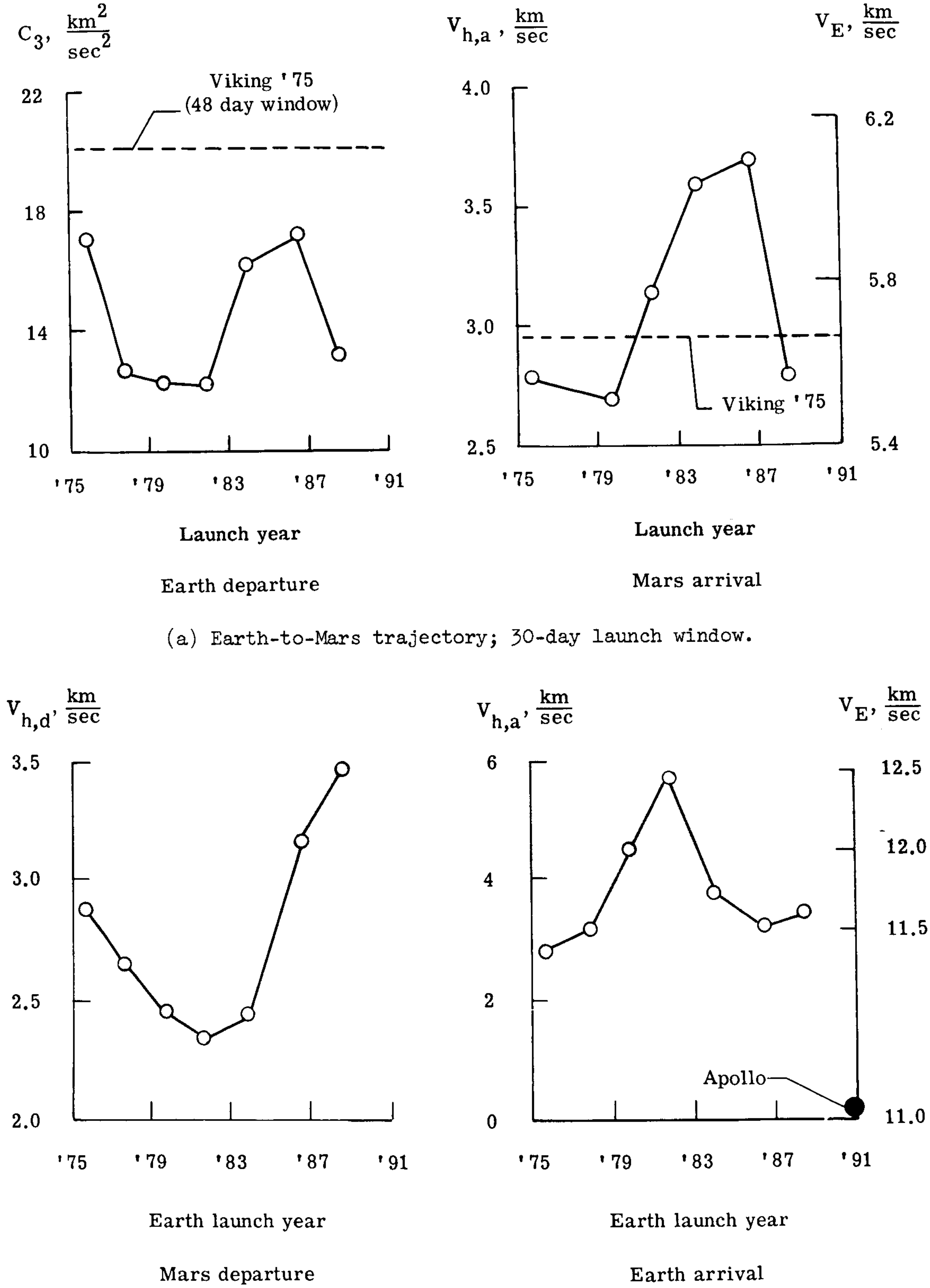
An analysis was performed to determine science requirements which produced the following important conclusions:

(1) A 50-g sample will be adequate and should contain at least some fines. The sample container will hold 200g of material, however, to insure extra material for verifying essential test results and conducting tests in other fields of interest.

(2) The sample should be acquired at a depth no greater than 5 cm and at a distance of at least 2.5 m from the lander center line.

CONTAMINATION BY PLANETARY ORGANISMS

All mission and systems designs meet the requirements of current policy regarding protection of Mars from Earth organisms. Most of the requirements on system design are met by applying techniques developed on the Viking mission. The problem of contamination of Earth by Mars organisms is assessed in a preliminary way. Systems for two missions, both employing Mars orbit rendezvous (MOR), are designed.



(a) Earth-to-Mars trajectory; 30-day launch window.

(b) Mars-to-Earth trajectory; 10-day launch window.

Figure 48.- Characteristic velocities for conjunction-class MSSR trajectories.

TRAJECTORY ANALYSIS AND DESIGN

A baseline trajectory profile was selected for its low energy requirements and relative simplicity of implementation. The baseline profile is distinguished by its employment of the Mars parking-orbit departure mode. Several trajectory profiles employing the MOR mode were investigated for comparison. In all, six different trajectory profiles were investigated. These profiles differ because of different modes employed for the Mars departure phase, the Mars entry phase, and the Earth arrival phase. Some pertinent conclusions follow:

(1) Efficient conjunction-class trajectories (interplanetary trajectories chosen for all missions) require about a year stay time at Mars. This requirement in combination with the requirement to limit stay-time on the surface led to missions employing Mars orbit-departure modes.

(2) The baseline trajectory profile imposes the greatest demands on the Mars entry systems and on the propulsion system of the Mars ascent vehicles (MAV).

(3) The MOR mode offers considerable flexibility in overall mission designs and, as a result, most of the missions investigated utilize MOR.

(4) The direct entry mode at Earth requires less energy and offers relative simplicity in implementation. The Earth orbit mode, however, offers more flexibility in design approaches, particularly in approaches to the problem of Earth contamination by Mars organisms.

SYSTEM DESIGN

Systems were designed for a total of twelve missions. Two missions employ the parking orbit departure mode at Mars, and ten employ the MOR mode. The constraints of the payload capability and payload compartment size of the Titan III E/Centaur were imposed on the system designs of most missions. Current technology was exploited to the greatest extent possible and systems were designed for compatibility with other missions and for ease of overall integration with other systems. Trajectory analysis was performed in the design studies for the Mars entry, Mars ascent, and Earth entry phases. Some pertinent conclusions follow:

(1) For the missions which impose the greatest demands on the Mars entry systems (baseline missions) sufficient mass can be entered without increasing the diameter of the aeroshell beyond that for the Viking '75 mission.

(2) Two-stage solid-propellant vehicles with high thrust-mass (T/M) ratios and spin-stabilized second stages make efficient Mars surface-to-orbit launch vehicles.

(3) The adoption of the spin-stabilized Pioneer 9 design for the Earth return vehicle and related systems, which was based on developed technology, led to a very efficient design for this key vehicle.

(4) Design concepts were developed for a data handling and command (DHAC) system which is based on existing technology but which is less massive and bulky than existing systems. This DHAC system is used extensively during the mission on several different vehicles.

MASS SUMMARY AND EARTH LAUNCH VEHICLE REQUIREMENTS

The resulting launch vehicle requirements are as follows:

(1) Ten of the missions investigated can be performed by using a single Titan III E/Centaur as the launch vehicle.

(2) All missions can be performed by using the Space Shuttle/Centaur as the launch vehicle.

(3) The baseline missions, although they require minimum energy, do not result in the minimum Earth launch mass.

(4) The missions requiring minimum Earth launch mass utilize the MOR mode, a MAV with a spin-stabilized second stage, and direct Mars entry.

(5) The missions requiring maximum Earth launch mass utilize the MOR mode, a MAV with a three-axis-controlled second stage, and an out-of-orbit Mars entry.

Langley Research Center,
National Aeronautics and Space Administration,
Hampton, Va., January 29, 1975.

APPENDIX A

DETAILED MASS BREAKDOWNS FOR SOME MAJOR VEHICLES AND MODULES

Summary mass tables are given in the appropriate sections of the paper for each of the major spacecraft components (except the bioshield) indicated in figure 9. The bioshield mass m is constant for all missions (110.7 kg). This appendix presents detailed mass breakdowns for each variation of the Mars cruise module, Mars orbit vehicle, Mars aeroshell, Mars parachute systems, Mars lander module, Mars ascent vehicle, and Earth return vehicle. (See tables XXV to XXXI.)

TABLE XXV.- DETAILED BREAKDOWN OF SYSTEMS AND MASSES FOR MARS CRUISE MODULE

[Asterisks denote totals whereas underlines denote subtotals]

System or component	Mass, kg
Structure:	
Bus	18.1
Propulsion support	1.8
Mars entry capsule and bioshield support	9.8
Solar panel substrates	8.7
Antenna and antenna support	3.2
Bracketry and fasteners	4.5
Electronics support structure	15.9
Solar panel support structure	4.5
Total	66.5
Telecommunications:	
Total based on Mariner 5	15.8
Power:	
Solar panels (less substrates)	4.4
Battery	6.2
Battery charger	0.7
Power switching and logic	1.4
2.4-kHz inverters (2)	1.6
400-Hz inverter	1.6
Boost regulators (2)	2.8
30-V dc regulator	1.9
Power distribution	1.1
dc/ac/dc converter/regulators	3.8
Total	25.5
Data handling and command	4.5
Pyrotechnics:	
Pyrotechnics control unit	2.0
Propulsion actuation unit	1.5
Pyrotechnic devices	1.9
Total	5.4

TABLE XXV.- DETAILED BREAKDOWN OF SYSTEMS AND MASSES FOR MARS CRUISE MODULE – Concluded

[Asterisks denote totals whereas underlines denote subtotals]

System or component	Mass, kg
Cabling:	
Intramodule cabling	21.3
Intermodule cabling	2.3
Total	23.6
Temperature control:	
Blankets	3.6
Louvers	3.4
Heaters	0.9
Total	7.9
Mechanical devices:	
Solar panel dampers	0.3
Antenna dampers	0.8
Solar panel deployment mechanisms	1.7
Separation initiated timer and pyro arming switch	0.4
MEC release mechanisms	2.1
Bioshield release mechanisms	2.2
Total	7.5
MCM – Less propulsion and attitude control	*156.7
Propulsion and attitude control:	
Electronics (Mariner 9)	15.3
Propulsion inerts (includes attitude thrusters and motors)	27.0
Propellant:	
20-m/s correction (spacecraft)	22.0
60-m/s deflection (MCM)	5.7
Total	70.0
Total MCM	**226.7

TABLE XXVI. - DETAILED BREAKDOWN OF SYSTEMS AND MASSES FOR MARS ORBIT VEHICLES

[Asterisks denote totals whereas underline denote subtotals]

System or component	Mass, kg				
	$(MOV)_1$	$(MOV)_2$	$(MOV)_3$	$(MOV)_4$	$(MOV)_5$
Structure:					
Bus	13.6	13.6	13.6	13.6	13.6
Attitude propulsion system support	1.8	1.8	1.8	1.8	1.8
MEC and bioshield support	9.5	9.5	9.5	9.5	9.5
Solar panel substrates	5.9	5.9	5.9	5.9	5.9
Antenna supports	0.5	0.5	0.5	0.5	0.5
Low-gain antenna structure	0.5	0.5	0.5	0.5	0.5
High-gain antenna structure	0.9	0.9	0.9	0.9	0.9
Bracketry and fasteners	4.6	4.6	4.6	4.6	4.6
Electronics support structure	9.0	9.0	9.0	9.0	9.0
Solar panel support structure	2.7	2.7	2.7	2.7	2.7
Docking cone support	2.7	2.7	2.7	2.7	2.7
Rendezvous and docking support ring	3.6	3.6	3.6	3.6	3.6
ERV support and guides	4.6	4.6	4.6	4.6	4.6
Total	59.9	59.9	59.9	59.9	59.9
Telecommunications:					
S-band radio (based on Mariner 5 design)	15.9	15.9	15.9	15.9	15.9
Antenna feeds, rotation joint	1.8	1.8	1.8	1.8	1.8
Total	17.7	17.7	17.7	17.7	17.7
Power:					
Same as Mars cruise module power system, (table XXV)	25.5	25.5	25.5	25.5	25.5
Additional solar cells	0.7	0.7	0.7	0.7	0.7
Less dc/ac/dc conversion	-2.6	-2.6	-2.6	-2.6	-2.6
Total	23.6	23.6	23.6	23.6	23.6
Data handling and command:					
Partial ("satellite") DHAC using ERV DHAC modules	1.8	1.8	1.8	1.8	1.8
Spinup/release event timer	0.5	0.5	0.5	0.5	0.5
Total	2.3	2.3	2.3	2.3	2.3

TABLE XXVI.- DETAILED BREAKDOWN OF SYSTEMS AND MASSES FOR MARS ORBIT VEHICLES – Continued

[Asterisks denote totals whereas underline denote subtotals]

System or component	Mass, kg				
	$(MOV)_1$	$(MOV)_2$	$(MOV)_3$	$(MOV)_4$	$(MOV)_5$
Pyrotechnics:					
Pyrotechnics control unit	2.7	2.7	2.7	2.7	2.7
Propulsion actuation unit	1.4	1.4	1.4	1.4	1.4
Pyrotechnic devices	3.2	3.2	3.2	3.2	3.2
Total	7.3	7.3	7.3	7.3	7.3
Cabling:					
Intramodule cabling	15.4	15.4	15.4	15.4	15.4
Intermodule cabling	3.2	3.2	3.2	3.2	3.2
Total	18.6	18.6	18.6	18.6	18.6
Temperature control:					
Blankets	10.0	10.0	10.0	10.0	10.0
Louvers	5.4	5.4	5.4	5.4	5.4
Heaters	0.9	0.9	0.9	0.9	0.9
Total	16.3	16.3	16.3	16.3	16.3
Mechanical devices:					
MEC release mechanism	2.3	2.3	2.3	2.3	2.3
Bioshield release mechanism	2.3	2.3	2.3	2.3	2.3
Solar panel deployment mechanism	1.4	1.4	1.4	1.4	1.4
Solar panel and antenna dampers	1.4	1.4	1.4	1.4	1.4
SIT and PAS	0.5	0.5	0.5	0.5	0.5
MOV/ERV cable release mechanism	0.5	0.5	0.5	0.5	0.5
Spin table and drive	3.6	3.6	3.6	3.6	3.6
ERV release mechanism	1.7	1.7	1.7	1.7	1.7
Docking cone release mechanism (including cable cutter)	1.7	1.7	1.7	1.7	1.7
Total	15.4	15.4	15.4	15.4	15.4
Articulation control:					
Solar panel actuators	3.6	3.6	3.6	3.6	3.6
Drive electronics	2.3	2.3	2.3	2.3	2.3
Total	5.9	5.9	5.9	5.9	5.9

TABLE XXVI.- DETAILED BREAKDOWN OF SYSTEMS AND MASSES FOR MARS ORBIT VEHICLES – Concluded

[Asterisks denote totals whereas underlines denote subtotals]

System or component	Mass, kg				
	$(MOV)_1$	$(MOV)_2$	$(MOV)_3$	$(MOV)_4$	$(MOV)_5$
Rendezvous and docking:					
Docking cone assembly	6.8	6.8	6.8	6.8	6.8
Approach guidance radar	6.8	6.8	6.8	6.8	6.8
GaAs light source assembly	0.9	0.9	0.9	0.9	0.9
X,Y optical trackers (2)	1.4	1.4	1.4	1.4	1.4
Approach guidance antennas (4) with feeds	1.8	1.8	1.8	1.8	1.8
Rendezvous and docking/DHAC/attitude propulsion system interface electronics	2.3	2.3	2.3	2.3	2.3
Total	20.0	20.0	20.0	20.0	20.0
MOV – less propulsion and attitude control	*187.0	*187.0	*187.0	*187.0	*187.0
Propulsion and attitude control:					
Electronics (Mariner 9)	15.3	15.3	15.3	15.3	15.3
Attitude control (inerts and propellant)	36.3	36.3	36.3	36.3	36.3
Propulsion inerts (includes engine)	114.3	114.3	114.3	453.6	340.2
Propellant for velocity changes:					
20-m/s midcourse correction	21.9	22.2	16.7	53.6	38.8
278-m/s orbit maneuver	56.4	59.1	56.4	87.5	76.0
1331-m/s orbit insertion	377.7	393.5	377.7	1888.6	1366.8
Total	621.9	640.7	616.7	2534.9	1873.4
Total MOV	**808.9	**827.7	**803.7	**2721.9	**2060.4

APPENDIX A - Continued

TABLE XXVII. - DETAILED MASS BREAKDOWN

FOR MARS ENTRY AEROSHELLS

System or item	Mass, kg				
	$(A/S)_1$	$(A/S)_2$	$(A/S)_3$	$(A/S)_4$	$(A/S)_5$
Structure:					
Struts, rings, propulsion supports	139	117	97	103	83
Thermal protection:					
Heat shield (phenolic nylon)	57	49	42	44	34
Internal insulation	9	9	9	9	9
Attitude-control system:					
12 thrusters, 2 tanks, lines, valves (for $(A/S)_4$ and $(A/S)_5$ enlarged propellant tanks and plumbing were used for the deorbit maneuver)	24	24	24	35	31
Roll control propellant	37	28	18	31	17
Deorbit propellant				74	49
Pressurant (helium)	2	2	2	4	3
Altimeter radar antenna	2	2	2	2	2
Science:					
Temperature, pressure sensor	1	1	1	1	1
Power transducers	1	1	1	1	1
Wiring	4	4	4	4	4
Aeroshell total	276	237	200	308	234

TABLE XXVIII.- DETAILED MASS BREAKDOWN FOR MARS PARACHUTE SYSTEMS

System or component	Mass, kg	
	$(P/S)_1$	$(P/S)_2$
Structure – truss	23	22
Base cover	43	39
Thermal protection:		
Base-cover heat shield	24	20
Internal insulation	6	5
Parachute:		
Pilot (D_o = 6.2 m), ribbon type	10	
Main (D_o = 22.9 m), DGB type	112	
Main (D_o = 16.1 m), (Viking)		43
Mortar:		
For pilot parachute	5	
For D_o = 16.1-m parachute		13
Main parachute canister (for D_o = 22.9 m)	6	
Pyrotechnics (end cap release)	1	1
Wiring	2	2
Total	232	145

APPENDIX A – Continued

TABLE XXIX.- DETAILED MASS BREAKDOWN

FOR MARS LANDER MODULES

Item	Mass, kg, for each entry-departure mode				
	$(MLM)_1$	$(MLM)_2$	$(MLM)_3$	$(MLM)_4$	$(MLM)_5$
Structure:					
Landing gear	77.1	57.1	27.2	57.1	27.2
Supports for electronics and RTG	18.1	18.1	18.1	18.1	18.1
Lower frame assembly	22.7	16.8	13.6	16.8	13.6
Sampler boom support	4.5	4.5	4.5	4.5	4.5
Sample processor/loader support	4.5	4.5	4.5	4.5	4.5
Descent propulsion supports	4.5	4.5	3.6	4.5	3.6
Intertruss	13.6	10.0	8.2	10.0	8.2
Aeroshell, parachute system separation rails	5.9	5.9	5.9	5.9	5.9
Fasteners, brackets	6.8	6.8	6.8	6.8	6.8
Total	157.7	128.2	92.4	128.2	92.4
Telecommunications:					
Radar altimeter	9.7	9.7	9.7	9.7	9.7
Radar altimeter antenna	0.4	0.4	0.4	0.4	0.4
Terminal descent and landing radar and antenna	21.5	21.5	21.5	21.5	21.5
S-band antenna (2) (low-gain, high-gain)	1.0	1.0	1.0	1.0	1.0
S-band receiver (2)	---	---	2.8	---	2.8
S-band traveling wave tube amplifiers (2)	---	---	0.9	---	0.9
S-band transmitter/oscillator/driver/module (2)	---	---	1.1	---	1.1
Diplexer (2)	---	---	0.6	---	0.6
Command detector (2)	---	---	1.1	---	1.1
Couplers, switches, feeds	0.5	0.5	0.7	0.5	0.7
S-band high-gain antenna pointing control	4.0	4.0	4.0	4.0	4.0
Total	37.1	37.1	43.8	37.1	43.8

TABLE XXIX.- DETAILED MASS BREAKDOWN

FOR MARS LANDER MODULES – Continued

Item	Mass, kg, for each entry-departure mode				
	$(MLM)_1$	$(MLM)_2$	$(MLM)_3$	$(MLM)_4$	$(MLM)_5$
Power:					
Radioisotope thermoelectric generator	15.6	15.6	15.6	15.6	15.6
Converters	1.6	1.6	1.6	1.6	1.6
Battery charger	1.3	1.3	1.3	1.3	1.3
Switching and control, sensors	0.7	0.7	0.7	0.7	0.7
Total	19.2	19.2	19.2	19.2	19.2
Data handling and command:					
Sampler command unit	0.5	0.5	0.5	0.5	0.5
Data handling unit	0.8	0.8	0.8	0.8	0.8
Total	1.3	1.3	1.3	1.3	1.3
Attitude control:					
Axial accelerometer (Total)	1.8	1.8	1.8	1.8	1.8
Pyrotechnics:					
Control unit	3.4	3.4	3.4	3.4	3.4
Propulsion actuation	1.5	1.5	1.5	1.5	1.5
MAV releases	0.6	0.6	0.6	0.6	0.6
Aeroshell releases	0.6	0.6	0.6	0.6	0.6
Parachute releases	0.6	0.6	0.6	0.6	0.6
Terminal descent tank releases	0.6	0.6	0.6	0.6	0.6
Landing leg releases	0.6	0.6	0.6	0.6	0.6
Tubing cutter	1.4	1.4	1.4	1.4	1.4
Total	9.3	9.3	9.3	9.3	9.3
Thermal control:					
Radioisotope thermoelectric generator windshroud	2.7	2.7	2.7	2.7	2.7
Radioisotope heat units (10)	1.4	1.4	1.4	1.4	1.4
Thermal switch	2.7	2.7	2.7	2.7	2.7
Insulation	13.6	13.6	13.6	13.6	13.6
Radioisotope thermoelectric generator coolant system	2.3	2.3	2.3	2.3	2.3
Total	22.7	22.7	22.7	22.7	22.7

APPENDIX A – Continued

TABLE XXIX. - DETAILED MASS BREAKDOWN

FOR MARS LANDER MODULES – Concluded

Item	Mass, kg, for each entry-departure mode				
	$(MLM)_1$	$(MLM)_2$	$(MLM)_3$	$(MLM)_4$	$(MLM)_5$
Wiring	10.3	10.3	10.3	10.3	10.3
Mechanisms:					
S-band antenna deployment	6.1	6.1	6.1	6.1	6.1
MAV release	2.7	2.7	2.7	2.7	2.7
Descent tank/MAV release	1.4	1.4	1.4	1.4	1.4
Landing gear/MAV release	1.4	1.4	1.4	1.4	1.4
Total	11.6	11.6	11.6	11.6	11.6
Propulsion:					
Engines	70.8	47.2	47.2	47.2	47.2
Lines, valves, connectors	13.2	11.8	11.8	11.8	11.8
Tanks	38.6	21.8	12.7	27.2	14.5
Pressurant	15.0	7.3	6.4	13.2	6.4
Residual propellant	10.0	5.9	3.6	7.3	3.6
Usable propellant	200.0	114.8	65.3	142.9	68.0
Roll thrusters	4.5	3.6	3.6	3.6	3.6
Total	352.1	212.4	150.6	253.2	155.1
Science:					
Triaxial accelerometer	1.1	1.1	1.1	1.1	1.1
Leg force sensors	0.7	0.7	0.7	0.7	0.7
Sampler boom and head	11.4	11.4	11.4	11.4	11.4
Sample sealing assembly	7.4	7.4	7.4	7.4	7.4
Sample transporter	3.2	3.2	3.2	3.2	3.2
Sample processor/loader	4.5	4.5	4.5	4.5	4.5
Sample temperature sensor	0.1	0.1	0.1	0.1	0.1
Sample scoop load sensor	0.2	0.2	0.2	0.2	0.2
Ambient light photometer	0.2	0.2	0.2	0.2	0.2
Anemometer – three-axis	0.4	0.4	0.4	0.4	0.4
Atmosphere temperature sensor	0.3	0.3	0.3	0.3	0.3
Humidity	0.3	0.3	0.3	0.3	0.3
Total	29.8	29.8	29.8	29.8	29.8
Total MLM	652.9	483.7	392.8	524.5	397.3
MLM without propulsion	300.8	271.3	242.2	271.3	242.2

TABLE XXX.- DETAILED MASS BREAKDOWN FOR MARS ASCENT VEHICLES

[Asterisks denote totals whereas underlines denote subtotals]

(a) Baseline mission

System or component	Mass, kg
First stage:	
Structure:	
Rocket-motor supports	9.1
Attitude-control electronics and battery support structure	0.9
Attitude-control system support and side panels	5.4
Attitude-control system propellant/pressurant tankage support	0.9
Upper frame assembly	9.1
Heat shield (jettison after 3 sec)	2.4
Ascent shroud	2.3
Total	30.1
Power:	
Battery (Ag-Zn, 18 cells, 10 A-hr, 280 W-hr)	6.1
Fuses, diodes	0.1
Total	6.2
Pyrotechnics:	
Propulsion actuation unit	1.1
Propellant isolation valve	0.2
Pressurant line-purge valve	0.2
Propellant line-cutting guillotines (3)	1.4
Gyros/attitude-control electronics/battery release devices	0.5
Thrust vector control thrusters/lower truss V-band	0.6
Rocket motor separation devices	1.3
Ascent shroud/upper ring/tankage separation devices	0.6
Rocket engine igniters with safe/arm circuits (3)	2.3
Lower heat-shield release devices	0.5
Total	8.7
Cabling:	
Intramodule cabling (primarily attitude-control system)	0.8
Intermodule cabling (primarily power and pyrotechnics)	1.1
Total	1.9

TABLE XXX. - DETAILED MASS BREAKDOWN FOR MARS ASCENT VEHICLES - Continued

[Asterisks denote totals whereas underlines denote subtotals]

(a) Baseline mission - Continued

System or component	Mass, kg
Mechanical devices:	
Gyros/attitude-control electronics/battery release mechanism (V-band)	0.4
MLM release mechanism	0.5
Upper frame release fittings and separation hardware	2.3
Rocket motor release and separation hardware (2 V-bands)	1.8
Thrust vector control assembly release and separation hardware (lower truss V-band)	0.9
Total	5.9
Temperature control	0.9
Attitude control (without propulsion):	
Rate gyros (3) with electronics	0.9
Wide-angle gyros (3) with electric heaters, ac converters	4.3
Attitude-control electronics	2.7
Two-axis pendulous sensor	0.2
Pitch/yaw thrusters (3)	5.4
Roll thrusters (2)	1.6
Plumbing, valving, transducers	14.1
Total	29.2
Attitude control (with propulsion):	
Propellant	11.3
Pressurant	0.4
Tankage	3.2
Total	14.9
Primary propulsion:	
Propellant	874.0
Inerts	116.1
Total	990.1
Total first stage	*1087.9

TABLE XXX.- DETAILED MASS BREAKDOWN FOR MARS ASCENT VEHICLES – Continued

[Asterisks denote totals whereas underlines denote subtotals]

(a) Baseline mission – Concluded

System or component	Mass, kg
Second stage:	
Structure:	
Rocket motor support	2.7
Pyrotechnics control unit support	0.5
Total	3.2
Pyrotechnics:	
Pyrotechnics control unit (for ERV propulsion isolation valve, ERV antenna release, MAV second-stage separation and ignition, all MAV first-stage separation events)	3.4
Second-stage separation devices	0.6
Second-stage igniter with safe/arm circuit	1.4
Total	5.4
Cabling:	
Intramodule and intermodule pyrotechnics and sensor cabling	0.5
Mechanical devices	0.4
Propulsion:	
Propellant	247.1
Inerts	33.3
Total	280.4
Total second stage	*289.9
Total MAV	**1377.8

TABLE XXX.- DETAILED MASS BREAKDOWN FOR MARS ASCENT VEHICLES - Continued

[Asterisks denote totals whereas underlines denote subtotals]

(b) MOR mission, three-axis-controlled second stage

System or component	Mass, kg
First stage:	
Structure:	
Rocket-motor support truss tubes	4.5
Rocket-motor support fittings	4.5
Upper frame assembly	9.1
Ascent shroud	2.3
Lower heat shield	2.7
Total	23.1
Pyrotechnics:	
Rocket engine igniters (3)	2.3
Rocket motor separation devices	1.4
Ascent shroud and upper ring separation devices	0.9
Lower heat-shield release devices	0.4
Total	5.0
Cabling:	
Pyro and sensor cabling (Total)	0.5
Temperature control:	
Plume shields for MAV (Total)	0.4
Mechanical devices:	
MLM release mechanism (residual)	0.4
Upper frame release fittings and separation hardware	2.3
Rocket motor V-bands (2)	1.8
Total	4.5
Propulsion:	
Propellant	594.2
Inerts	79.8
Total	674.0
Total first stage	*707.5

TABLE XXX.- DETAILED MASS BREAKDOWN FOR MARS ASCENT VEHICLES – Continued

[Asterisks denote totals whereas underlines denote subtotals]

(b) MOR mission, three-axis-controlled second stage – Continued

System or component	Mass, kg
Second stage:	
Structure:	
Bus and electronics support	10.0
Attitude propulsion system thruster support and side panels	5.4
Attitude propulsion system propellant and pressurant tankage support	0.9
Canister boom support	1.4
Bracketry and fasteners	1.8
Antenna support structure	0.9
Main MOV engine support structure	2.7
Solar panel support structure	1.8
Solar panel substrates	4.5
Total	29.4
Telecommunications:	
Same systems as total on baseline ERV (see table XXXI)	7.2
S-band low-gain antenna	0.4
S-band high-gain antenna	0.9
Total	8.5
Power:	
Solar panel cell assemblies (75 W at 1.6 AU, 73 W required prior to rendezvous maneuver)	4.5
Battery (18 cell Ag-Zn, 10 A-hr, 280 W-hr)	6.1
Battery charger	0.7
Power distribution	1.1
Fuses, diodes	0.2
Telecommunications/DHAC/traveling-wave tubes/regulators	3.6
Miscellaneous	1.4
Total	17.6

TABLE XXX. - DETAILED MASS BREAKDOWN FOR MARS ASCENT VEHICLES – Continued

[Asterisks denote totals whereas underlines denote subtotals]

(b) MOR mission, three-axis-controlled second stage – Continued

System or component	Mass, kg
Data handling and command:	
Same DHAC as on baseline ERV (see table XXXI)	4.5
Pyrotechnics:	
Pyrotechnics control unit	1.8
Propulsion actuation unit	1.4
Separation devices for first stage	1.8
Main MOV engine igniter (with safe/arm)	1.4
Total	6.4
Cabling:	
Intramodule cabling	5.0
Intermodule cabling	1.8
Total	6.8
Temperature control:	
Insulation	2.7
Louvers	1.8
Heaters	0.9
Total	5.4
Mechanical devices:	
Canister boom with drive head and separation mechanism	3.6
Solar panel dampers	0.4
Canister insertion ram	0.9
Solar panel deployment mechanisms	1.8
Total	6.7
Attitude control (includes propulsion):	
Sensors	8.6
Thrusters	4.1

TABLE XXX. - DETAILED MASS BREAKDOWN FOR MARS ASCENT VEHICLES – Continued

[Asterisks denote totals whereas underlines denote subtotals]

(b) MOR mission, three-axis-controlled second stage – Concluded

System or component	Mass, kg
Plumbing, valving, transducers	14.1
Tankage	3.2
Pressurant	0.5
Propellant	11.3
Total	41.8
Rendezvous and docking (Total)	5.0
Primary propulsion:	
Propellant	203.2
Motor	27.2
Total	230.4
Contingency mass	1.4
Total second stage	363.9
Total MAV	**1071.4

TABLE XXX.- DETAILED MASS BREAKDOWN FOR MARS ASCENT VEHICLES – Continued

[Asterisks denote totals whereas underlines denote subtotals]

(c) MOR mission, spin-stablized second stage

System or component	Mass, kg
First stage:	
Structure (Total)	16.8
Power – same as $(MAV)_1$, Stage 1 (Total)	6.2
Attitude control:	
Total (less tankage, pressurant, propellant)	20.6
Tankage, pressurant	2.3
Propellant	8.6
Total	31.5
Pyrotechnics and mechanical devices (Total)	4.5
Cabling (Total)	3.8
Temperature control (Total)	4.1
Primary propulsion:	
Motor	40.8
Propellant	314.0
Total	354.8
Total first stage	*421.7
Second stage:	
Structure (Total)	11.3
Power:	
Battery	6.1

TABLE XXX.- DETAILED MASS BREAKDOWN FOR MARS ASCENT VEHICLES – Concluded

[Asterisks denote totals whereas underlines denote subtotals]

(c) MOR mission, spin-stabilized second stage – Concluded

System or component	Mass, kg
Distribution	0.8
Miscellaneous	2.0
Total	8.9
Telecommunications:	
Total (less antennas)	7.3
Antennas	1.3
Total	8.6
Pyrotechnics and mechanical devices (Total)	5.4
Temperature control (Total)	3.2
Rendezvous (Total)	4.1
Data handling and command (Total)	4.5
Propulsion:	
Main motor	10.4
Propellant main motor	73.3
Kick motors and propellant	2.3
Despin motors and propellant	1.8
Total	87.8
Contingency	1.4
Total second stage	*135.2
Total MAV	**556.9

TABLE XXXI. - DETAILED MASS BREAKDOWN FOR EARTH RETURN VEHICLES

System or component	Mass, kg		
	$(ERV)_1$	$(ERV)_2$	$(ERV)_3$
Structure:			
Bus (frame, side panels, equipment platform)	8.8	8.8	8.8
Bracketry and fasteners	1.6	1.6	1.6
Attitude propulsion system support structure	0.9	0.9	0.9
ERV main engine support structure	1.4	1.4	1.4
EEC support	0.5	0.5	0.5
Antenna	0.9	0.9	0.9
Total	14.1	14.1	14.1
Telecommunications:			
S-band receivers (2)	2.8	2.8	2.8
S-band traveling wave tubes (2)	0.9	0.9	0.9
Transmitter oscillator/driver/modulator (2)	1.1	1.1	1.1
Diplexers (2)	0.6	0.6	0.6
Driver-to-traveling wave tube coupler	0.2	0.2	0.2
Coaxial radio frequency switches (2)	0.2	0.2	0.2
Antenna feeds	0.4	0.4	0.4
Command detector units (2)	1.0	1.0	1.0
Total	7.2	7.2	7.2
Power:			
Solar cell array	5.4	5.4	5.4
Battery (3 A-hr, Ni-Cad)	5.4	5.4	5.4
Battery charger	0.7	0.7	0.7
dc/ac/dc converter/regulators (2)	1.6	1.6	1.6
Power switching and logic	0.9	0.9	0.9
Traveling-wave tube power converters (2)	2.1	2.1	2.1
Intrasubsystem wiring	0.5	0.5	0.5
Total	16.6	16.6	16.6

TABLE XXXI. - DETAILED MASS BREAKDOWN

FOR EARTH RETURN VEHICLES – Continued

System or component	Mass, kg		
	$(ERV)_1$	$(ERV)_2$	$(ERV)_3$
Data handling and command:			
Central timers (2)	0.3	0.3	0.3
Command processors (2)	0.8	0.8	0.8
Control memories (2)	0.8	0.8	0.8
Command output units (2)	0.5	0.5	0.5
Mode control and sequencer units (2)	0.4	0.4	0.4
Analog multiplexers (2)	0.5	0.5	0.5
Analog-to-digital converters (2)	0.2	0.2	0.2
Input/output control and digital formatter units (2)	0.5	0.5	0.5
Data memories (2)	0.2	0.2	0.2
Interface circuit units (2)	0.3	0.3	0.3
Total	4.5	4.5	4.5
Pyrotechnics:			
Pyrotechnics control unit (including capacitance banks)	1.4	1.4	1.4
Propulsion actuation unit	1.3	1.3	1.3
Helium press, initiation valve	0.2	0.2	0.2
ERV main engine igniter and safe/arm circuit	1.4	1.4	1.4
EEC release device	0.3	0.3	0.3
Antenna release device	0.2	0.2	0.2
Total	4.8	4.8	4.8
Temperature control:			
Insulation (including ERV main engine nozzle cowling)	2.7	2.7	2.7
Louvers	1.4	1.4	1.4
Heaters (Sun sensor, transmitter, oscillator, propellant line)	0.4	0.4	0.4
Total	4.5	4.5	4.5

TABLE XXXI. - DETAILED MASS BREAKDOWN

FOR EARTH RETURN VEHICLES – Concluded

System or component	Mass, kg $(ERV)_1$	$(ERV)_2$	$(ERV)_3$
Mechanical devices:			
Antenna deployment mechanism	0.5	0.5	0.5
EEC separation mechanism	0.6	0.6	0.6
MAV second-stage separation mechanism	0.9	0.9	0.9
Nutation damper	1.3	1.3	1.3
Total	3.3	3.3	3.3
Cabling:			
Intramodule cabling (hardwired) and intermodule cabling (with connectors)	3.2	3.2	3.2
Attitude control (no ΔV propulsion)	14.3	14.3	14.3
Contingency	1.1	1.1	1.1
ERV – less propulsion	*73.6	*73.6	*73.6
ΔV propulsion:			
Propellant	14.1	5.8	6.0
Tankage	3.2	2.3	2.7
Total	17.3	8.1	8.7
Propulsion for Mars departure:			
Propellant	69.1	56.3	58.5
Inerts	9.4	9.4	9.4
Total	78.5	65.7	67.9
Total ERV (without bioshield)	**169.4	**147.4	**150.2
Bioshield			9.1
Total ERV (with bioshield)			**159.3

APPENDIX B

SEQUENCE OF MISSION OPERATIONS

The sequence of mission operations is as follows:

Baseline Mission (Missions 1 and 2 From Table VI)

Earth departure and Earth-to-Mars cruise:

(1) Earth departure
(2) Spacecraft separation
(3) Establish two-way communications (between MCM and Earth control)
(4) Sun acquisition
(5) Canopus acquisition
(6) Maintain three-axis stabilized attitude
(7) Separate bioshield cap
(8) Trajectory determination
(9) Midcourse maneuver

Mars approach and entry:

(1) Trajectory determination and correction as required
(2) Transmit descent program (store in ERV command memory)
(3) Perform MEC checkout sequence
(4) Enable all required subsystems in MEC
(5) Separate MEC
(6) MCM deflection
(7) Turn on radar altimeter
(8) Deploy pilot parachute
(9) Deploy main parachute
(10) Separate aeroshell
(11) Deploy landing gear
(12) Turn on terminal descent and landing radar
(13) Initiate descent retropropulsion
(14) Touch down
(15) Terminate and seal retropropulsion

Surface operations:

(1) Establish landed mode (including two-way communications)
(2) Playback descent data
(3) Transmit surface science data
(4) Acquire surface sample

(5) Perform sample processing/loading sequence
(6) Perform inner sample canistering operation
(7) Transport inner canister to outer canister shell (located in EEC or EOC)
(8) Braze-seal inner canister lid to outer canister shell
(9) Initiate canister pressure/temperature monitoring
(10) Prepare for ascent
(11) Receive and store (on ERV) ascent program

Mars surface to orbit launch:

(1) Enable required MAV/ERV subsystems
(2) Transmit launch command
(3) Initiate stored ascent program
(4) Unlatch MAV
(5) Ignite MAV stage 1 motors
(6) Lift off
(7) Separate lower heat shield
(8) Command roll and pitch thrusters
(9) Stage 1 motor burnout
(10) Separate stage 1 motors
(11) Reorient MAV for stage 2 firing
(12) Spin up MAV
(13) Separate remainder of stage 1 MAV
(14) Initiate Mars orbit injection maneuver
(15) Ignite stage 2 motor
(16) Separate stage 2 after motor burnout
(17) Enable ERV attitude/propulsion subsystem
(18) Orient ERV for apoapsis kick
(19) Perform apoapsis kick (to raise periapsis)
(20) Orient ERV (spin stabilized) for normal orbital operations
(21) Initiate ERV operations (including two-way communications)

Mars orbiting:

(1) Orbital cruise (telemetry transmitted briefly per Earth control command)
(2) Orbit determination
(3) Orbit trim as required
(4) Post-trim orbital cruise

Mars departure and Mars-to-Earth cruise:

(1) Receive and store transfer trajectory command program
(2) Orient ERV for Earth injection
(3) Ignite ERV main motor

(4) Reorient for cruise
(5) Trajectory determination
(6) First midcourse maneuver
(7) Reorient for cruise
(8) Trajectory determination
(9) Second midcourse maneuver

Earth arrival and entry:

(1) Pre-entry ERV orientation
(2) EEC separation and EEC beacon turn-on
(3) EEC ballistic deceleration
(4) Deploy parachute (pressure-actuated switch)
(5) Impact surface

Earth orbit alternate:

(1) Pre-injection ERV orientation
(2) Separate EOC
(3) EOC retromotor burn

MOR Missions (Missions 3 to 12 of Table VI)

Earth departure and Earth-to-Mars cruise:

(1) Same sequence as for baseline mission

Mars approach and orbit insertion (direct entry):

(1) Trajectory determination and correction as required
(2) Transmit descent program (stored in MAV command memory)
(3) Perform MEC checkout sequence
(4) Enable all required subsystems in MEC
(5) Separate MEC
(6) Insert MOV into orbit
(7) Separate back cap bioshield

Mars approach and orbit insertion (out-of-orbit entry):

(1) Trajectory determination and correction as required
(2) Propulsively brake MEC and MOV into orbit
(3) Separate MEC from MOV
(4) Separate back cap of bioshield
(5) Perform MEC deorbit maneuver

Mars entry and surface operations:

(1) Same sequence as for baseline mission

Phase-one orbiting:

(1) Maintain two-way communications with Earth control center

(2) Check out rendezvous equipment

Mars surface-to-orbit launch:

(1) Initially same sequence as on baseline mission

(2) Reorient MAV for stage 2 motor burn

(3) Initiate Mars orbit injection sequence

(4) Ignite MAV stage 2 motors

(5) Stage 2 motor burnout

(6) Deploy solar panels

(7) Establish orbital cruise mode (solar power, two-way communications link)

(8) Orient MAV for apoapsis kick

(9) Perform apoapsis kick (raise periapsis)

(10) Orient MAV (three-axis stabilized) for rendezvous operations (reestablish orbital cruise mode after every maneuver)

Orbit rendezvous:

(1) Determine MAV and MOV orbits (command MOV/maneuvers as required)

(2) Activate approach guidance radar on MOV (at a distance of 30 km between vehicles)

(3) Perform rendezvous up to station-keeping point (distance of 10 m for three-axis MAV, MOV, 3 m for spinner MAV)

(4) Earth-based status check (both vehicles)

Docking and sample transfer (MAV with three-axis-controlled second stage):

(1) Initiate docking mode (turn on modulated light source on MAV; X,Y optical trackers on MOV; switch MOV guidance to optical trackers; MOV performs final maneuvers toward MAV)

(2) Dock and latch

(3) Transfer sample canister assembly from MAV to EEC or EOC and attach canister compartment double-cap assembly

(4) Verify canister installation complete (via MOV telemetry data)

Alternate sample transfer for MAV with spin-stabilized second stage:

(1) Initiate modulated light source on MAV

(2) Initiate X,Y optical tracking on MOV

(3) MOV performs precision alinement with MAV

(4) Despin MAV

(5) Low-velocity ejection of sample canister from MAV

(6) Capture of canister by EEC or EOC

(7) Attach sample canister by braze seal

(8) Signal completion of transfer and sealing

Undocking and ERV separation:

(1) Separate docking adapter (to which MAV is latched) from MOV (for three-axis MAV only)

(2) Shut down MAV power

(3) Reorient MOV for ERV separation and proper ERV cruise orientation

(4) ERV spinup

(5) ERV separation

(6) Separate ERV bioshield and initiate EOC seal plug combustion (for Earth backcontamination mode only)

(7) Deploy ERV solar panels and acquire Sun and stellar attitude reference

(8) ERV antenna deployment

(9) Reestablish two-way communications link with ERV (now using ERV antenna)

Mars departure through end of mission:

(1) Same sequence as baseline mission

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